Final Report
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Technical Report

STUDY OF DESIGN PARAMETERS OF EXPLOSIVE INITIATORS
WITH RESPECT TO SPACE ENVIRONMENTS

by

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THE FRANKLIN INSTITUTE RESEARCH LABORATORIES

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ABSTRACT

The purpose of this study has been to obtain the latest information on environment to be expected in Manned Space trips as far away as Mars and to relate this information to the expected effects on explosive devices in such a way that potential problems with explosive devices can be anticipated and corrected and realistic specifications can be developed for explosive devices which take into consideration such environmental effects.

The study has considered the duration of space flights and the report includes discussions on such environmental factors as temperature, pressure, electromagnetic, electrical, particle, chemical and shock and vibration. Consideration is given to earth orbiting vehicles and to trips to Venus, Mars, the Moon and the largest, nearest asteroids.

Within these boundaries it is concluded that trips into the Venus atmosphere could represent some serious problems because of predicted high temperature and pressures although there is still much to learn about the Venus atmosphere.

In addition to this specialized set of conditions, the study indicates possible areas of concern due to long term temperature exposures in space, possible electrostatic charge build up on the spacecraft; impact by particulate matters and long range effects due to some of the present sterilization procedures.

It is concluded that while there is no environmental problem, other than possibly the Venus atmosphere situation, that is so serious as to disrupt any present programs, that initiation of programs at this time can reduce the severity of future problems and improve the reliability of explosive devices.

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1. FOREWORD

Many definitions of explosives and pyrotechnics can be found, but considered from a user's viewpoint, they are convenient sources of large amounts of mechanical and thermal energy, releasable almost instantaneously upon demand. It is because of this that they are of value to armed forces and to miners, quarrymen, construction engineers and space explorers. In relatively small packages, explosives and pyrotechnics perform tasks such as operating switches or valves, opening bolted joints, rupturing separating membranes, driving studs, igniting rocket motors, starting fuel pumps, separating missile and rocket stages, and many others.

Because of their small size and weight, and the small amount of triggering energy required, they are extensively used in missiles and spacecraft to perform necessary functions. The most commonly used explosive devices in this area are actuated electrically, and are therefore called electroexplosive devices, or more commonly EEDs.

Over the next decade or two, manned spacecraft are expected to be used on a variety of missions which could produce environments possibly intolerable to explosive devices of present design and materials. Since both safety and reliability of the explosive devices are of prime importance in these missions, the design of future devices must be suitable for all of the possible conditions to which they might be exposed.

Because information useful for explosive devices in extra-terrestrial environments is both scant and scattered, NASA Manned Spacecraft Center engaged the Franklin Institute Research Laboratories (Contract NAS 9-4042) to gather information on the possible environmental conditions to which manned spacecraft missions may be subjected within the next ten to twenty years and to analyze this information for application to the design of explosive initiators which will be used on these spacecraft.

2. INTRODUCTION

2.1 Scope of Study

It was agreed that the study should provide broad background material regarding the environments to which a manned spacecraft initiator and its related systems will be exposed. We are concerned with all characteristics - chemical, electrical, thermal, magnetic, or any others which conceivably influence the initiators and their circuits; we are also concerned with the compatibility of materials and design concepts of spacecraft initiators with these environments. The study is not concerned with environments or conditions prior to launch unless such factors have a definite effect after launch. For example, spacecraft are usually coated with chemical sprays prior to launch, therefore, many residual chemicals may be carried on the spacecraft after launch. Other than such special cases, the direct interest of the study is in launch, total flight, and termination of flight, whether on earth or other bodies. The study is limited to the following manned missions:

a. Trips to Mars
b. Trips to Venus
c. Lunar laboratories
d. Earth-orbiting laboratories
e. Trips to the largest nearby asteroid.

Guides for the direction of effort were also adopted as follows:

a. We are concerned with both external and internal (on-board) environments.

b. While of secondary consideration, the various range requirements must be kept in mind during the study since it will be necessary to launch from these ranges.

c. Design guides or recommendations should be general rather than specific in detail. The final initiator specification will probably be written not only from the results of this program but also from several others which are underway. This study is to produce not the specifications, but background information for the specifications.

d. Where there is an obvious need for information, limited scope tests can be conducted to clarify the situation. If this lack can be filled only by a more extensive program, we will recommend such programs.

e. Each of the firing systems is to be considered in its entirety with regard to the environments.

f. Commonality is desired for all manned spacecraft systems.

g. It is desirable that EEDs have the same external dimensions as the present Apollo Standard Initiator.

h. The duration of the missions is an important factor.

It is also to be noted that the study is to cover explosives in general, not merely electroexplosive devices. But because by far the greater quantity of explosive is in EEDs, and also because the EEDs have, generally, more sensitive explosive components, most of the study will have to do with these devices.

2.2 Program Plan

To provide a systematic study plan, the work was divided into specific technical groups.

a. Duration of Exposures: An estimate of the times, on the various type, for which there would be exposures to the several hazards.

b. Thermal and Barometric

c. Electromagnetic radiation environments: to include effects caused by sun spot activity, the Van Allen Belt and solar storms, as well as the frequencies and magnitudes associated with specific planets, man made environments and electrical noise.

d. Other electrical environments and effects: to include electrostatic effects, lightning here and on other planets, and transients or steady state signals from on-board equipment.

e. Particle environments: to include the density of occurrence, size, velocity, and energy of atomic and nuclear particles as well as meteorites and micrometeorites.

f. Shock and vibration: to include the magnitudes and frequencies of vibration, shocks and acceleration factors.

g. Chemical and biological environments: to include sterilization procedures, planetary environments, bacteria and moisture effects.

h. Materials: to include explosives and related material, such as can be made available for use under the environmental conditions in the systems, within limits of compatibility with other materials likely to be used.

3. RESULTS OF STUDY

3.1 Duration of Exposure

It is readily understood that the effect of any environment is a function not only of the nature of the environment but also of the length of time over which the exposure occurs. Table 1 gives one way and round trip times, excluding orbit and landing times and based (except as noted) on a consensus from many reports.

TABLE 1

ESTIMATED TIME IN EARTH DAYS, MISSION TRANSIT

	Min	Max	Typical
Moon (one way)	2	3	2.8
(round trip)	4.5	7	6.5
Venus (one way)	146	312	150
(round trip)	-	-	440
Mars (one way) [(1)]	207	253	-
(round trip) [(1)]	325	560	-

Notes: For deep penetration trips such as Mars or Venus, the date of launch has a strong bearing on total transit time. These figures assume the selection of a favorable date.

The time of a Mars round trip tends toward the lower figure if in-flight propulsion is used, and if aerodynamic lift in the Mars atmosphere is used.

Most of the elapsed time of a mission is spent in interplanetary space. The time spent in the vicinity of the destination, that is, within the environments associated with the destination, depends on specific flight details. Based on numerous sources, Tables 2, 3, and 4 give an estimated breakdown for ambient temperature and pressure exposures, for typical flight.

(1) See numbered references at end of report.

Table 2

LUNAR MISSION: TEMPERATURE-PRESSURE EXPOSURE
(Typical Maximum Values)

Spacecraft Position	Launch To Parking Orbit	Parking Orbit	In Transfer	At Lunar Surface
Duration	9 minutes	37 minutes	3 days	as scheduled
Temperature	216 to 293°K	125 to 430°K	125 to 430°K	100 to 400°K
	-57 to 20°C	-148 to 157°C	-148 to 157°C	-173 to 127°C
	-70 to 60°F	-234 to 315°F	-234 to 315°F	-280 to 260°F
Pressure (torr)	7.6×10^{2} to 3.0×10^{-10}	3.0×10^{-10}	1.0×10^{-13}	7.6×10^{-11}

Table 3

MARS MISSION: TEMPERATURE-PRESSURE EXPOSURES
(Estimated Maximum Values)

Spacecraft Position	Launch to Parking Orbit	In Parking Orbit	In Transfer	Entering Mars Atmosphere	At Mars Surface
Duration	109 minutes	40 minutes	250 days	6 minutes	As scheduled
Temperature	216 to 293°K	125 to 430°K	125 to 430°K	140° to 303°K	170 to 310°K
	-57 to 20°C	-148 to 157°C	-148 to 157°C	-133 to 30°C	-103 to 37°C
	-70 to 68°F	-234 to 315°F	-234 to 315°F	-211 to 86°F	-155 to 97°F
Pressure (torr)	7.6×10^{2} to 3.0×10^{-10}	3.0×10^{-10}	1.0×10^{-13}	1.0×10^{13} to 76	76

Table 4

VENUS MISSION: TEMPERATURE-PRESSURE EXPOSURES
(Estimated Maximum Values)

Spacecraft Position	Launch to Parking Orbit	In Parking Orbit	In Transfer	Entering Venus Atmosphere	At Venus Surface
Duration	10 minutes	40 minutes	200 days	150 minutes	As scheduled
Temperature	216 to 293°K -57 to 20°C -70 to 68°F	125 to 430°K -148 to 157°C -234 to 315°F	125 to 430°K -148 to 157°C -234 to 315°F	200 to 700°K -73 to 427°C -100 to 800°F	315 to 700°K 42 to 427°C 106 to 800°F
Pressure (torr)	7.6×10^{2} to 3.0×10^{-10}	3.0×10^{-10}	1.0×10^{-13}	2.0×10^{-6} to 4.5×10^{4}	4.5×10^{4}

3.2 Thermal and Barometric Environments

Basing estimates on the best information available, and using all applicable references consulted in this task, we find that atmosphere pressures and temperatures vary to extremes. The last columns in Tables 2,3, and 4 give specific examples. In most cases relatively large spreads are indicated, and in the case of Venus, there is still considerable controversy regarding actual values.

The temperature listed for these atmospheres could become serious problems for most initiators. The upper limit of over 400°C far exceeds the 75°C (165°F) standard most initiators must meet; the lower limit of -148°C, likewise, is much beyond the specified -54°C (-65°F). Most initiators are presently designed to function in very low pressures, and proper design for most of the conditions listed here should be within present technology. The atmosphere of Venus, however, may have pressure 60 times as great as the earth atmosphere, and initiators are rarely checked for functioning under such large external pressure. Such a pressure would be roughly equivalent to that at a depth of 2000 feet in the Earth oceans. Additional thought should be given to these problems, which could become very serious.

In empty space there is no heating or cooling by convection or conduction. Nevertheless, radiation may have a strong effect upon the temperature of the vehicle. This will be discussed under "Electromagnetic Radiation."

3.3 Electromagnetic Radiation Environment

The major source of electromagnetic radiation in our solar system is the sun. The average energy output is remarkably constant; however, it is well known that transient disturbances, such as solar flares, release substantial amounts of energy. Observations have shown that these levels may be a million times greater than the quiet emission. The chart in Figure 1 (adapted from Ref. 2) represents the latest data on maximum spectral irradiance of the sun from 10^{-4} microns to 10^{7} microns. For convenience of analysis, we have divided the spectrum into the following bands.

1. Radio frequencies - 1×10^{3} to 1×10 microns
2. Optical(Infrared to Ultraviolet) - 1×10^{-2} to 1×10^{3} microns
3. X-Rays - less than 1×10^{-2} microns

Each of these divisions is treated in a different manner in the following sections. Because the band of all wavelengths longer than 10^{8} microns (100 meters) contains a negligible amount of energy, it can be ignored.

3.3.1 Radio Frequencies (1×10^{3} microns to 1×10^{8} microns)

The radio emission of a large solar flare is on the order of 10^{24} ergs (approximately equivalent to 2.8×10^{13} watt-hours). Even though a typical flare may have an area of 10^{15} square meters, we can consider it as a point source when viewed from the earth (which is 1.5×10^{11} meters from the sun). This means that the 10^{24} ergs are radiated in a spherical pattern. With this energy spread over a spherical surface having a radius of 1.5×10^{11} meters, we would expect the energy density on our planet to be very small.

To avoid the problem of atmosphere absorption and to make the values usable in outer space, the calculations are made for the top of our atmosphere. The astrophysical unit for spectral irradiance is erg/cm^2-sec-micron.* This can be converted to electrical power units, since 1 aiu = 10^{-3} watts/meter^2 - micron.

*We will use the abbreviation "aiu" for astrophysical irradiation unit.

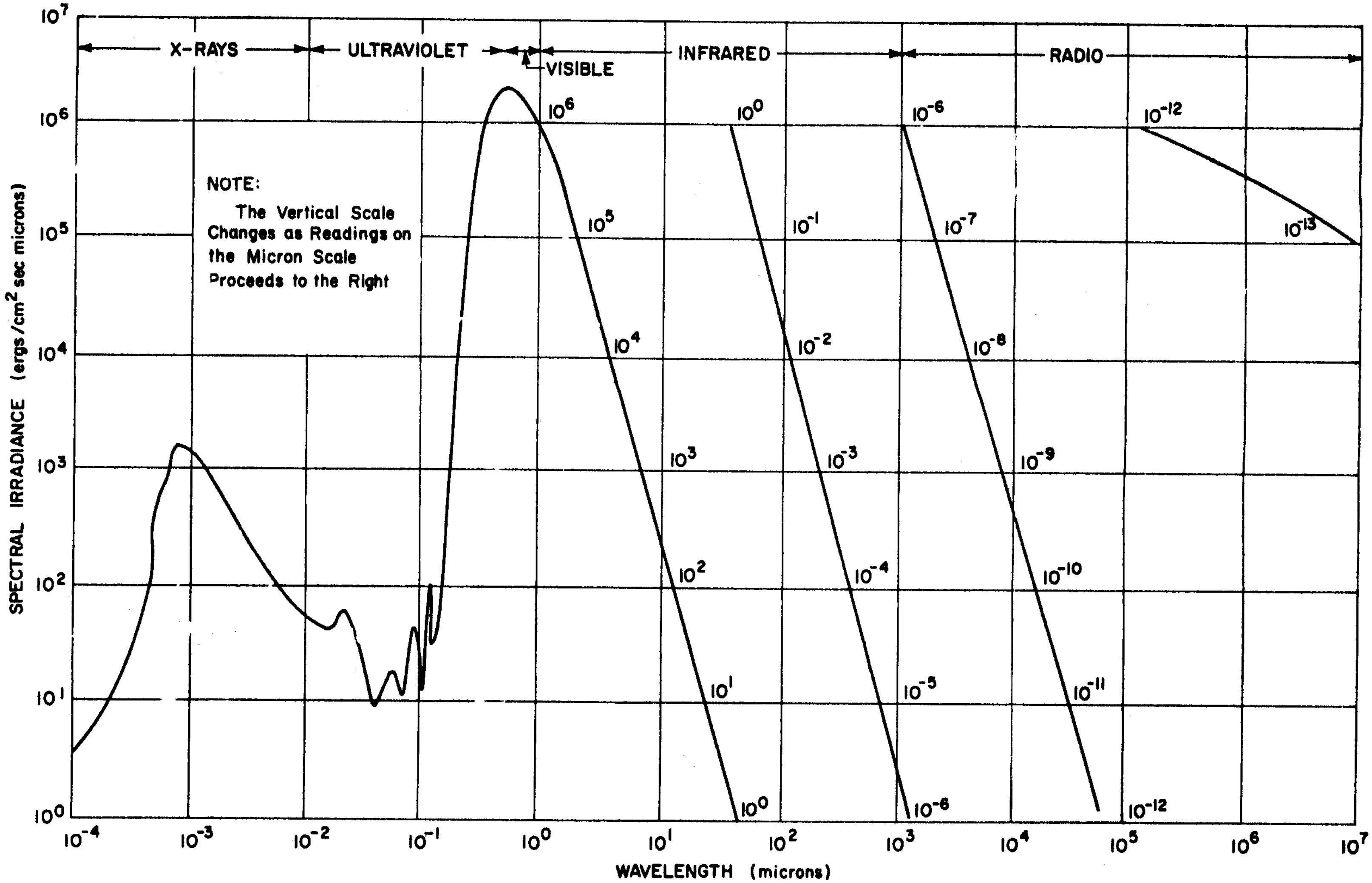


FIG. I. THE SOLAR SPECTRUM

The latest estimates indicate a level of 10^{-6} aiu at 300 gigacycles which drops to 10^{-13} aiu at 3 megacycles. If we were to make the worst case assumption of irradiance of 10^{-6} aiu over the total radio spectrum (300 gigacycles to 3 megacycles, or 1×10^{3} microns to 1×10^{8} microns) we could then compute the total power density as follows:

$$P_T = (10^{-6}\ \text{aiu}) \times (1 \times 10^{5}\ \text{microns}) = 10^{-2}\ \text{ergs/cm}^{-2}\ \text{sec} = 10^{-4}\ \text{watts/m}^2$$

This means that if all of the energy in the radio spectrum impinging on an antenna with an aperture of one square meter could be completely transferred to a load, there would be only 0.1 milliwatt. We would conclude from these data that the radio spectrum emitted by the sun (both steady and bursts) is not hazardous to electroexplosive circuits at a distance from the sun corresponding to Earth's position.

If the spacecraft approaches the sun it will intercept more energy. The Eastern Test Range has set 100 watts per square meter as the maximum level that the vehicle can be exposed to (on the launching pad) for safety to personnel and ordnance devices [(3)]. Let us take one watt per square meter as a conservative maximum that we would tolerate in space and calculate how close we could approach the sun without exceeding this value.

$$\frac{P_1}{P_2} = \frac{x^2}{D^2}$$

$$\frac{1 \times 10^{-4}}{1} = \frac{x^2}{(1.5 \times 10^{11})^2}$$

$$x = 1.5 \times 10^{9}\ \text{meters}$$

where

P_1 = power density at top of earth's atmosphere (watts/m^2)

$P_2 = 1 \text{ watt/m}^2$

x = distance from surface of sun for 1 watt/m^2 (meters)

D = distance from surface of the sun to earth (meters)

The closest planet to the sun is Mercury with a distance of 5.8×10^{10} meters. To pick up one watt per square meter we calculated that we would have to be within 1.5×10^9 meters of the sun or 56.5×10^9 meters inside the orbit of Mercury. On a flight from Earth to Mercury there would probably be no reason to go this far inside the orbit; therefore, we do not consider radio frequency energy as a hazard to electroexplosives used on a vehicle in space even for a prolonged period of time.

3.3.2 Optical (1×10^{-2} microns to 1×10^3 microns)

When we move from the radio spectrum into the infrared visual-ultraviolet (optical) spectrum, we no longer talk about power intercepted by an antenna but now refer to the energy absorption of the skin of the vehicle, and its effect on the temperature of the spacecraft. This approach appears to be well accepted [4,5,6,7].

The main problem in predicting the temperature of a vehicle in space is knowing the radiative properties of the outer skin. Because of the absence of air around the vehicle, conduction and convection are negligible; this leaves radiation as the sole means of heat transfer.

In environments where heat transfer to or from a surface is entirely or primarily through radiation, a knowledge of the surface absorptance α and the surface emittance ε is necessary for heat-transfer calculations. There are conditions, however, where only the ratio of absorptance to emittance α/ε need to be known to calculate the temperature or performance of a system; the individual values of α and ε need not be used. These are conditions where the internal heat fluxes to or from the surface are either zero or small compared with the external, radiant heat fluxes; these conditions apply to a space vehicle exposed to sunlight. In this case, α becomes the total directional absorptance of solar radiation α_s.

Gray surfaces are those for which both the monochromatic absorptance α_λ and the monochromatic emittance ϵ_λ are invariant with wavelength. For perfectly diffuse surfaces, $\alpha_\lambda = \epsilon_\lambda$, and therefore, for gray diffuse surfaces, $\alpha = \alpha_\lambda = \epsilon_\lambda = \varepsilon$ and $\alpha/\varepsilon = 1.0$. This is not the case for most materials. Values of α_s/ε for metals are greater than unity because their monochromatic absorptance and emittance decrease with increasing wavelenght λ. Therefore, α_s is greater for the relatively short wavelength radiation from the Sun than is ε for the longer wavelength radiation emitted from the much colder surface. Semiconductors are similar to metals in that their α_s/ε is greater than 1. Nonconductors have α_s/ε values less than unity because α_λ and ϵ_λ increase with increasing wavelength for this class of compounds.

It would appear that α could not be greater than ε without violating Kirchhoff's law of radiation which states:

"The monochromatic emissivity (ε) of a surface at temperature T, is equal to its monochromatic absorptivity (α)" [8].

A discussion of this paradox is given in reference 4. The abstract of the article is given below.

"In applying Kirchhoff's law to calculate the emission of a heated body radiating freely to the outside, one assumes that the emission of a body when radiating freely is the same as its emission when enclosed in a blackbody cavity. But the radiation field is everywhere smaller in the first case than in the second; consequently the emission arising from induced radiative transitions must be smaller for a freely radiating body than the emission arising from these transitions when the body is enclosed in a cavity. Since the emissions differ, one is apparently led to the conclusions that Kirchhoff's law cannot be valid for a freely radiating body. It is shown that this conclusion is false: Kirchhoff's law is valid as long as the distribution over material states is the equilibrium distribution, and is therefore, in this sense, independent of the state of radiation field. One must, however, take proper account of the effect of induced emission in calculating the absorption coefficient". Table 5 gives the ratio of absorptance to emittance, α/ε, for several materials[5].

TABLE 5

RATIO OF ABSORPTANCE TO EMITTANCE FOR VARIOUS MATERIALS

Material	α/ε
Copper, polished	4.8
Aluminum, iridized and baked	3.6
Aluminum, shiny	1.5
Aluminum, iridized	1.2
Solar cells, silicon, glass coated	1.1
Black wrinkle finish	1.1
Black paint	1.0
Rokide A	0.3

Equilibrium temperature of a body suspended in free space is given by[6] Equation 1

$$T = \left[\frac{C}{4\sigma} \frac{\alpha_s}{\varepsilon}\right]^{1/4} \tag{1}$$

where:

T = temperature of irradiated surface in °K

C = solar constant, 0.135 watts/cm^2 (*)

ε = total emittance of irradiated surface

α_s = total directional absorptance to solar radiation

σ = Stefan-Boltzmann constant, $5.673 \times 10^{-12} \frac{W}{cm^2 \, °K^4}$

*The solar constant,[16] or the total energy at all wavelengths received from the sun at the top of the earth's atmosphere, amounts to 1.94 calories per square centimeter per minute (0.135 watts per square centimeter). It is represented by the total area under the curve of the solar spectrum. Ninety-eight percent of this total energy lies within the part of the spectrum we are considering; therefore, 0.135 watts per square centimeter is a reasonable value to use.

EXAMPLES

Case I. If the vehicle skin is copper, for which $\alpha/\varepsilon = 4.8$, then

$$T = \left[\frac{C}{4\sigma} \cdot \frac{\alpha_s}{\varepsilon}\right]^{1/4}$$

$$T = \left[\frac{0.135}{4(5.673 \times 10^{-12})} \cdot (4.8)\right]^{1/4}$$

$$T = 411°K \text{ or } 138°C$$

Case II. If the vehicle skin is aluminum, for which $\alpha/\varepsilon = 1.2$, then

$$T = \left[\frac{C}{4\sigma} \cdot \frac{\alpha_s}{\varepsilon}\right]^{1/4}$$

$$T = \left[\frac{0.135}{4(5.674 \times 10^{-12})} \cdot (1.2)\right]^{1/4}$$

$$T = 290\ °K \text{ or } 17\ °C$$

From these results it can be seen that the choice of skin material is important. Spraying the surface with a nonconductor, for which $\alpha/\varepsilon < 1$, would reduce the equilibrium temperature even more. It is interesting that, the values in Table 5 substituted in Equation 1 were used to predict the temperatures in the TIROS weather satellite and the calculated values agreed with the measured values [5].

Several sources of data are available to indicate that the preceding method is valid. One such source of data is the excellent "Problems and Research Effort in Space Environment Effects", by Gerhard B. Heller, paper #36 in Reference 9, which contains the chart shown in Figure 2. The dotted lines represent the two values of α_s/ε, 4.8 and 1.2, which we used in Equation 1 to calculate equilibrium temperature. Very close agreement is obtained.

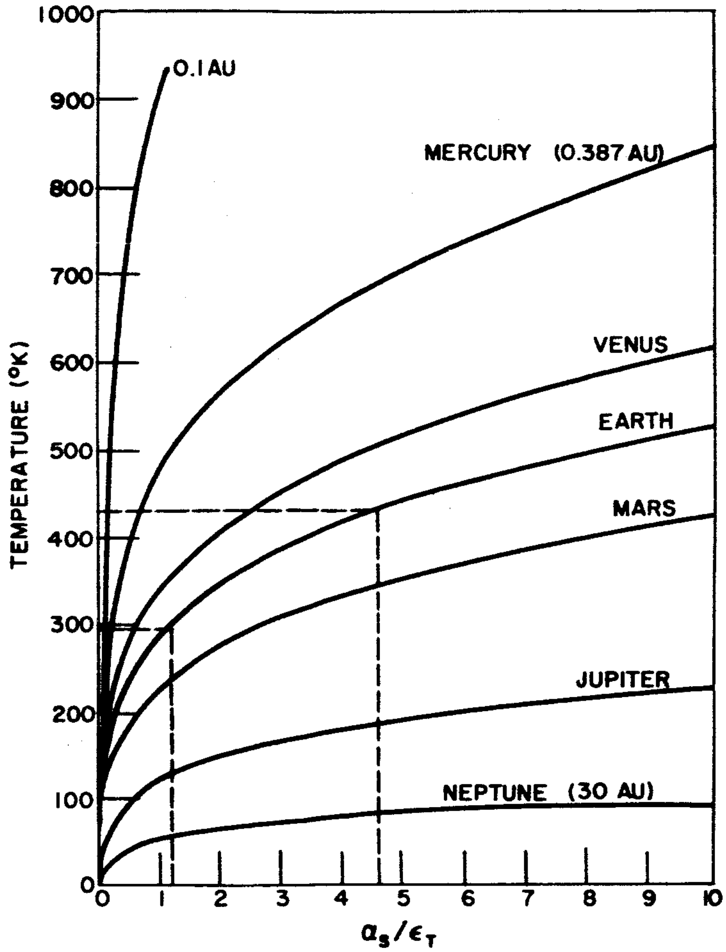


FIG. 2. TEMPERATURE OF SPACECRAFT AT VARIOUS DISTANCES FROM THE SUN

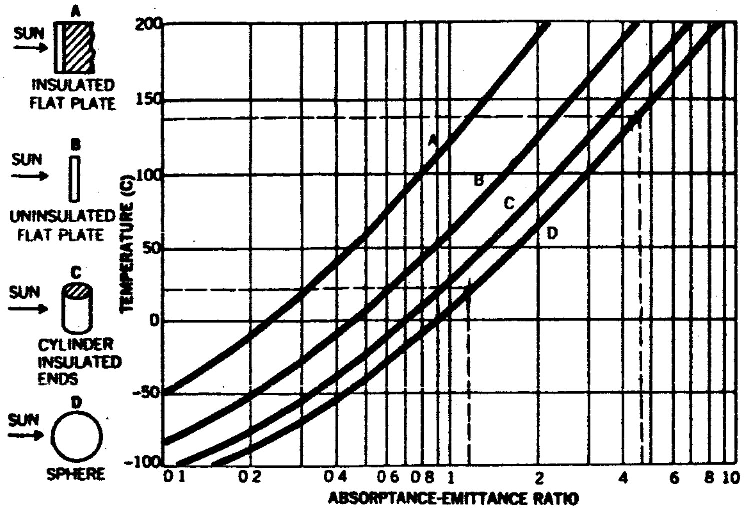


FIG. 3 TEMPERATURE OF VARIOUS SHAPED BODIES IN SPACE

Another chart [10] showing temperature equilibrium is given in Figure 3. An interesting point that this chart illustrates is that the shape of the free body influences the final temperature. The smaller the area that is perpendicular to the incident rays of the Sun, the less the energy absorbed. A sphere is the best (coolest) shape. Because a spacecraft will not be spherical its shape must be taken into account.

We are here concerned with the effect of this heating on explosives, whether they be in the explosive device or as a propellant. Let us take a look at Mariner IV, which has successfully travelled from Earth to within 7000 miles of Mars, and see how the temperature was controlled. For this vehicle active* as well as passive* temperature control was used. Movable louvers actuated by temperature sensitive bimetal elements are used to change the type of surface exposed to the sun. From the data we have it appears that temperatures below 35°C can be achieved without a great deal of difficulty.

*Passive control refers to a fixed condition, such as coating the spacecraft with a material that has a low α_s/ε ratio, to maintain a stable low temperature. Active control means a system that can vary the heat balance as required, such as by changing the α_s/ε ratio or by turning on a heating element.

Paper #54 of Reference 9, "Effect of Absorptance - Emittance Ratio on the Storage of Cryogenic Propellants in Space", delas with the possibility of carrying liquid hydrogen or liquid oxygen on long flights. To prevent vaporization these must be kept at temperature of about -250°C. A combination of surface temperature control and heat transfer control (insulation) is used to prevent temperature rise at the tanks. Considering the wealth of information that we have on the heating effect of solar radiation, we would conclude that it need not be a hazard to explosive components.

3.3.3 X-Rays (less than 1×10^{-2} microns)

The remaining portion of the solar radiation spectrum for consideration is the x-ray region. We may ignore the non-solar radiation since all measurements made so far indicate that the intensity of x-rays originating outside our solar system are many magnitudes below that emanating from the sun.

The bulk of the steady radiated energy in the solar spectrum lies between the wavelengths of 0.3 and 4 microns, with approximately 1% of the energy lying beyond each of these limits(11). Solar flares, however, can be a problem. In the x-ray spectrum (0.1 Å - 100 Å or 10^{-5} to 10^{-2} microns), flares may cause the incident radiation intensity in outer space to increase four orders of magnitude over background intensity levels(12). Fortunately, the x-ray output of flares lasts only for minutes or fractions thereof. In our analysis we will consider the peak values of flux intensity as shown by Figure 4 and will always assume complete absorption.

The order of magnitude of the activation energy or free energy change associated with many reactions is in the vicinity of 10^5 cal/mole, or 1.6×10^{-19} cal/molecule, or 4 eV/molecule. Since a photon energy of 4 eV corresponds to a wavelength of about 3000 Å, irradiation of wavelengths shorter than that will produce reactions(13).

Although we have never heard of x-rays damaging explosives, and very little information of EED sensitivity to x-rays is available in the literature, yet it is conceivable that x-rays may degrade EEDs* or even cause pre-ignition; therefore consideration of this hazard in outer space (vacuum) is warranted. But x-rays are known to damage semiconductors; and if we make the safe assumption that explosives are less sensitive, we can use semiconductors as our reference. If they are not affected, then we can assume that an EED would likewise be unaffected. Semiconductor materials, such as silicon or germanium,

*We will discuss electroexplosive devices (EEDs) since they usually contain the most sensitive explosive material; however, data can apply to any explosive material.

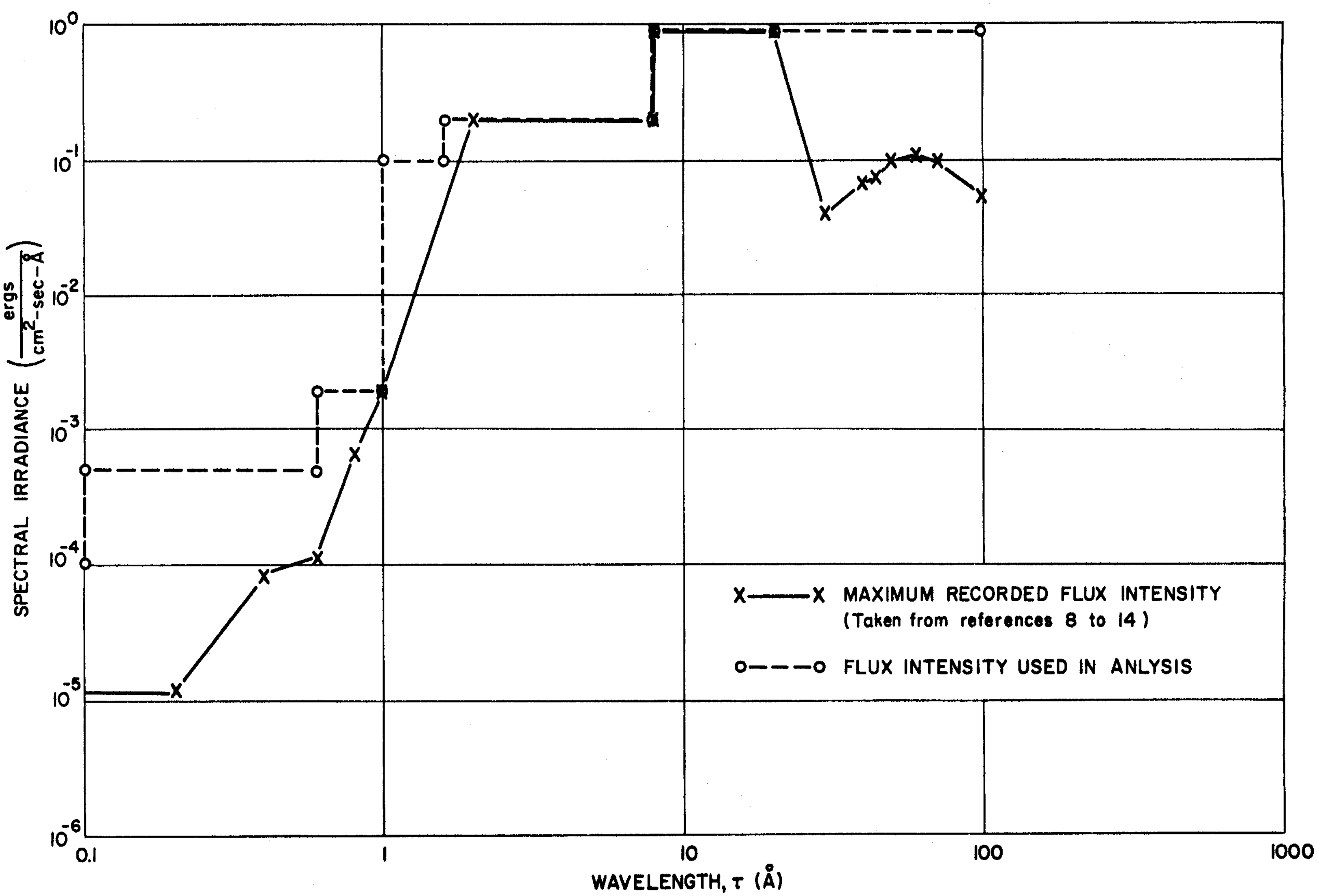


FIG. 4. SPECTRAL IRRADIANCE OF X-RAYS IN SPACE

are among the most sensitive to damage resulting from bombardment by high-energy particles, x-rays and gamma rays and would, therefore, appear to be excellent for the purpose. We chose silicon transistors for experimental use, because they are less resistant to radiation than germanium transistors.

Typically, the forming of an interstitial-vacancy defect pair in silicon requires imparting a threshold energy from 15 to 30 eV to the recoiling atom. This in turn implies a minimum energy for the incident particle, which is approximately 150 keV for electrons[14].

Ordinarily one thinks of radiation being measured in roentgens; however, this is not correct terminology when absorption is being considered[15]. A clear distinction should be made between the exposure dose measured in roentgens and the absorbed dose measured in rads. The exposure dose is measured in terms of the ability of the radiation to produce ionization in a given amount of air. The absorbed dose is measured in terms of the energy that the incident radiation can impart to the absorbing material. Since we are interested in solid material rather than air, the rad is the unit of interest.

One rad is defined as 100 ergs/g; however, three conditions must be specified when using this unit:

1. type of radiation
2. type of material being irradiated
3. amount of material being irradiated

Without these three factors the data would be meaningless since the same dose at a different wavelength could produce different results.

When discussing radiation damage to semi-conductors, it is common to use carbon as the reference material. This practice will be followed in this report.

Damage thresholds for semiconductor devices, depending on their sensitivity, fall within 10^5 to 10^7 ergs/gram of carbon (1000 to 100,000 rads).

We will assume 10^5 ergs/gram of carbon as the damage level. Other component parts and materials have their thresholds a little less than 10^9 ergs/gram of carbon, indicating that degradation other than that caused by radiation damage should be of chief concern[16].

Exposures in vacuum to x-rays of about 0.38 Å, carried out on forsterite (Alsimag-243), alumina, and steatite showed little changes even in surface electrical properties. BeO also showed little change electrically, but changed color from white to dark gray. Such a change could be significant where emissivity is important. Investigators of the effects of x-ray radiation often find that apparent changes in resistivity, dissipation factor, and dielectric constant of inorganic insulators are within the range of instrument error[17].

Data of space x-ray radiation from solar flares indicate that a maximum flux intensity of 0.9 ergs/cm^2 sec occurs between 8 Å and 20 Å[18]. If we assume that a transistor is 1 mm^3 or larger, and was irradiated at 0.9 ergs/cm^2-sec from 0.1 Å to 100 Å for 4 minutes this transistor would approach its damage threshold. Two factors must be noted. The first is this high flux intensity is not distributed generally from 0.1 Å to 100 Å, but only from 8 Å to 20 Å. Also, it is not always present in space but may appear occasionally for only a few seconds. This makes our assumed flux to be at least 100 times greater than the practical hazard. The second is shielding, which has yet to be considered.

If we assume that the vehicle has a metal skin, the explosive device will have some shielding. Also, the explosive is probably enclosed in a metal case, offering additional shielding. It can be shown, however, that at some particular wavelength or small wavelength band, metals may offer little or no attenuation to x-rays. Fortunately, the wavelengths where negligible attenuation occurs are below 1 Å, where the flux intensity is relatively small (less than 2×10^{-3} ergs/cm^2 sec). Aluminum offers less resistance to x-rays than copper, iron, or lead. We will, therefore assume in our analysis that aluminum is the shielding material. A 1/16 inch aluminum sheet offers attenuation by a factor of greater than 10^2 to x-rays at 1 Å and 10^{45} at 12 Å. At longer wavelengths x-ray attenuation is no problem. These calculations are based on the exponential attenuation formula:

$$I = I_o e^{-ux}$$

where

I_o = incident flux intensity
I = transmitted flux intensity
u = absorption coefficient
x = absorber thickness

Considering flux intensity and 1/16 inch aluminum shielding, our EED will still be safe after 5 years of continuous irradiation at the levels shown in Figure 4. Again, we stress that these flux levels are the recorded maximums in outer space, and under normal conditions we may expect an EED to be safe even after 100 years of exposure.

In summary, the following assumptions, which maximize worst conditions, were used in our analysis of effects of x-rays.

1. Maximum flux intensity from known solar flares is taken to be the constant intensity level of space.

2. Continuous maximum irradiation (normal to the device) and total absorption are assumed.

3. 1/16 inch aluminum, with less x-ray attenuation than copper, iron, or lead, is used as the shielding metal.

4. Attenuation by the case of an initiator is neglected.

5. The explosive is taken to be as sensitive to x-radiation as silicon transistors; but the transistors are known to be much more readily affected than any explosive now used in missile work.

Even with all these worst-case assumptions, it was found that continuous irradiation for 5 years is needed before an initiator becomes noticeably damaged. This indicates that x-rays are not a problem in space to initiators.

3.4 Other Electrical Environments and Effects

There are numerous electrical phenomena that must be considered, other than electromagnetic radiation of an oscillatory nature (radio, light, x-ray) discussed in the previous section. These others include induction by motion in a magnetic field; static discharge; thermocouple effects; motion relative to charged particles (electrons, protons from nuclear radiation, solar flares, or other sources; and lightning discharges.

3.4.1 Conductor Motion in a Magnetic Field

The voltage induced in a conductor moving in a uniform magnetic field is given by $V = Bv\ell \times 10^{-8}$ where

V is the potential, volts

B is the flux, gauss

v is the component of velocity, cm/sec, perpendicular to the direction of flux

ℓ is the conductor length, cm

Velocities that can be expected in a Venus expedition can be as high as 50,738 feet per second and Mars expeditions will have velocities around 43,189 ft/sec.[19] These velocities, the greatest in the missions under consideration, will occur near the earth and upon re-entry to the earth. Because there is no readily ascertainable basis for doing otherwise, we shall make the worst-case assumption that the motion is perpendicular to the flux.

The magnetic field in interplanetary space is estimated[20] at 5 to 10 gamma (1 gamma = 10^{-5} gauss). Upon approach to the earth it is not expected that fields would exceed 100 gamma in the regions where these velocities could be maintained. In fact, it is more likely to be on the order of that found in free space; the earth's field contributes little at distances greater than a few hundred kilometers.

Assuming, however, that a field of 100 gamma could exist, the potential induced would be on the order of $1.52 \times 10^{-5} \times \ell$ volts. A maximum value of ℓ would be on the order of 10 feet or 304 cm. This yields a potential of 4.62×10^{-3} volts. With the Apollo standard initiators, which has a bridge resistance of one ohm, the current would be on the order of 4 milliamperes. This current would be sustained for a relatively short time and is not, by itself, considered troublesome.

We have learned in our investigation that measurements made of the magnetic fields in the vicinity of the planets Mars and Venus indicate that none of them appear to have a magnetic field of significant magnitude compared to that of the earth.

Data from Explorer VI[(21)] gave a picture of the magnetic nature of earth's surroundings using magnetic search coil in the spin-stabilized vehicle. The nature of this coil was such as to give an indication of the component of the magnetic field perpendicular to the spin-axis of the vehicle. The expected magnetic field computed prior to flight of Explorer VI was not always in close agreement with the measured values. The differences could be explained by a toroidal circulating current, at 5 to 7 earth radii. Other measurements show that such a current, with a density of about 10^{-8} amperes/square meter, exists at that distance.

The boundary between the geomagnetic field and the interplanetary gas is taken to be neutral hydrogen plasma with a high degree of ionization. The temperature is 10^{3} to 10^{5} degrees Kelvin, the electrical conductivity 10^{12} to 10^{14} esu (copper is 5×10^{17} esu); hydrogen ion density is $500/cm^{3}$. Measurements of the plasma show fields of less than 5000 gamma (1 gamma $= 10^{-5}$ gauss). These findings are of interest only as showing that the magnetic fields are small enough to be considered safe; they would not induce a dangerous amount of energy. The period of change of the field as a result of magnetic storms varies from a hundred seconds to several days, and these changes are on the order of 10^{-3} gauss.

The voltage induced as the result of an external magnetic field in a circuit aboard a spacecraft will be proportional to the rate of change of the magnetic field. For this reason fields changing in either space or time would be of particular interest in this analysis. While the information now available indicates slow rates of change and weak fields, it is of course impossible to state with certainty that there are no conditions of field concentration in which a very fast-moving vehicle might pick up a hazardous amount of energy. We do feel, however, that a hazard is very unlikely.

Venus is reported to have no magnetic field of its own.[22] Similar findings, not yet confirmed, are reported for the planet Mars as a result of measurements aboard the recent Mariner IV flight.

The magnetic effects during flight to Earth's Moon to Mars, and to Venus will have no appreciable influence on electroexplosive devices or on the explosives themselves. It is highly unlikely that any modification to this statement will be needed, unless measurements and calculations are inadequate or in error by an order of magnitude or more.

3.4.2 Static Discharge

Accumulation of Static Charge

Electrostatic charge accumulation and discharge is a real danger. It is known that some sensitive explosive devices are highly sensitive to static. Static charge accumulation may behave in two ways.

Under suitable circumstances the charge resulting from the static accumulation may produce a relatively steady current in the form of corona. The magnitude of such a current is normally less then a few milliamperes, and in general, this current will be relatively harmless to an EED if it passes through the bridgewire. Should such a current continue for an extended period, however, the possibility of cook-off of the EED explosive or degrading of the EED performance must be considered. It would be wise to choose one of the explosives that is relatively insensitive to such effects, when the bridgewire cannot be kept out of the discharge path.

On the other hand, the electrostatic charge can built up and then be released in a sudden surge of current or spark. If such current finds a path internal to the initiator, particularly if a discharge occurs between the pins and the case of an EED through the explosive, it might fire the initiator.

Studies indicate that a cloud of ice crystals in the lower atmosphere of earth is one source of electrostatic charge that may occur in space travel.

There are numerous examples of the sources and magnitudes of electrostatic charge buildup which may be useful. For example, measurements on conventional aircraft operating in the earth's atmosphere show potentials of 200 volts to be common although these potentials are often quickly dissipated by corona currents. In the upper atmosphere of this earth, charging currents produced by meteoric dust impinging on space probes have been observed to be in the range of 60 to 600 picoamperes. Once again such charges are usually quickly removed by leakage. In addition, charging currents from the effects of rocket motor exhaust have been measured at more than 150 microamperes for a 5-inch solid fuel rocket motor, but the total net charge has been difficult to determine because of charge leakage. Some charge can also be developed by particle impact; see Section 3.5.6.

Explosive Response to Static Discharge

From our study of pertinent literature we find that an accumulation of static charge on spacecraft could produce potentials of several hundred kilovolts on the spacecraft particularly if charging occurs within the earth's atmosphere. Potentials of this magnitude must certainly be considered as hazardous to explosive devices, and with this in mind we reviewed some of the literature on explosive sensitivity to sparks and electrostatic discharge.

It is the consensus of researchers in this field that three factors are important with respect to the sensitivity of explosives to electrical discharges: (1) the minimum spark energy required for ignition, (2) the electrical properties of the materials, and (3) the environment of the material. In attempting to find the minimum energy required to fire explosives, it was soon discovered that the sensitivities determined by a number of experimenters varied greatly. This variation was partly due to the fact that there was great variation in the methods of determining sensitivities.

One of the best sources of information[23] included data on a number of common explosive materials, which are summarized in Figure 4. The data were obtained by discharging various capacitors charged to 500 volts through the test material by means of a pointed electrode. The circuit used was similar to the one shown in Figure 6. The test material was placed on a metal plate and the pointed electrode was lowered until discharge occurred.

Other studies have been made, with a resistance added in series with the capacitor and gap. A sample of the material under study was placed within the gap, which was usually a pointed electrode over a flat metal plate, as described earlier. Some of the values reported from various experimenters in this areas are given in Table 6.[24] Particularly noteworthy in this table as well as in the text of the referenced report is the effect of adding series resistance to the firing circuit. For mercury fulminate, note that as the series resistance is added the energy required on the firing capacitor decreases. It has been observed that this is the rule rather than the exception for most explosive materials tested in this fashion, an exception being lead styphnate, which requires the least energy with no series resistance. A general rule has been developed relating series resistance, capacity and minimum energy. It states that the smaller the capacity, the lower the minimum energy and the greater the optimum series resistance. The minimum energy requirements for ignition of an explosive material are changed by several orders of magnitude in some instances by the resistance in series with a circuit. Mr. A.R. Boyle in his thesis at the University of Birmingham, United Kingdom, (1943) showed a reduction in the ignition energy of mercury fulminate from 0.5 joule without any series resistance to 0.00375 joules with a series resistance of from 250,000 to 750,000 ohms.

This information pertains to circuits containing electroexplosive devices as well as to explosively loaded components, and is applicable in the pin-to-case testing of electroexplosive devices, or in the bridgewire to bridgewire testing of these devices for static safety. It has been previously pointed out that the human circuit is not the only means by which static energy can be delivered to explosive devices

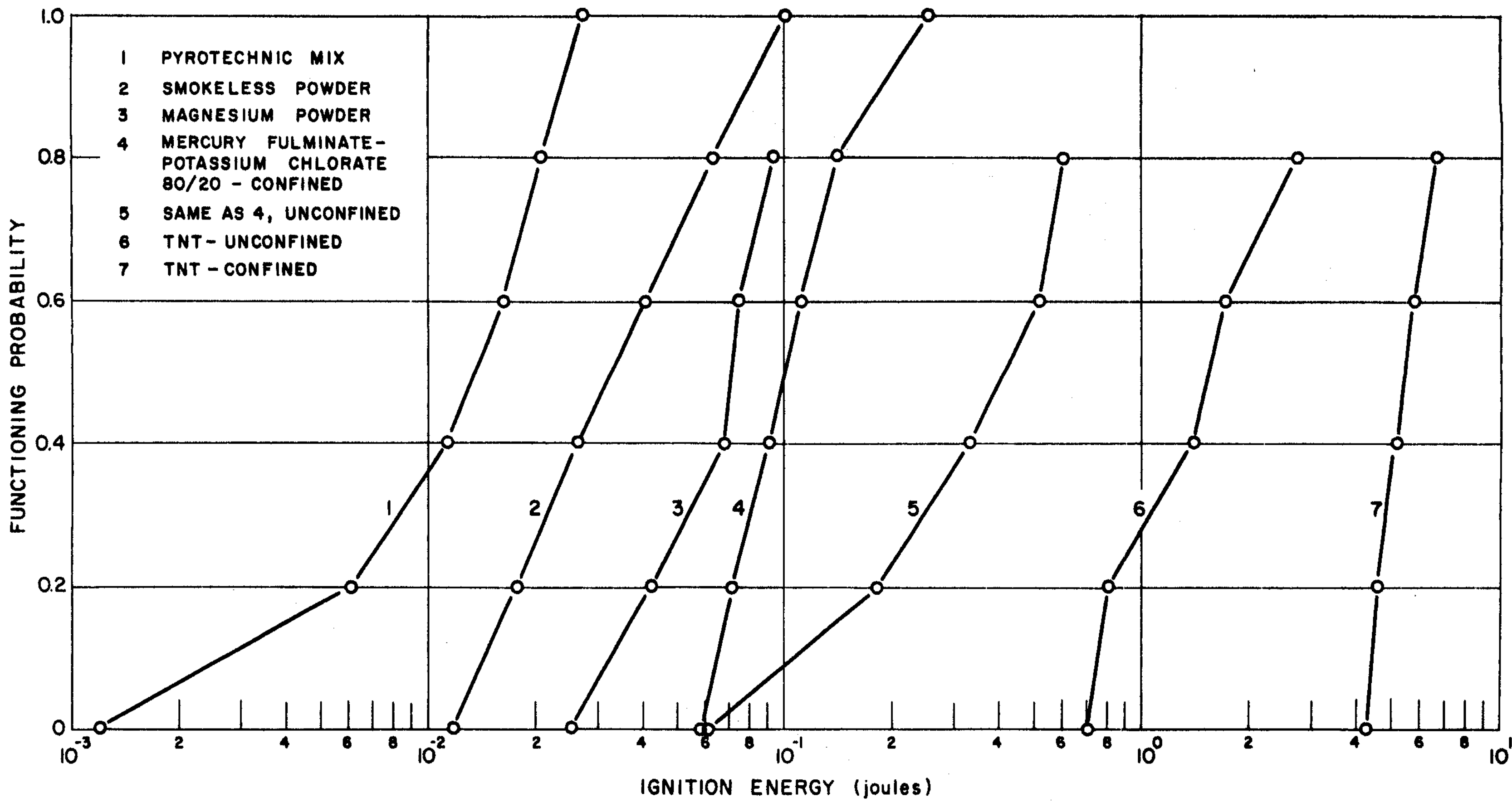


FIG. 5. ENERGY REQUIRED FOR GIVEN FUNCTIONING PROBABILITY OF COMMON EXPLOSIVE MATERIALS

(Data extracted from Bureau of Mines investigation 5002)

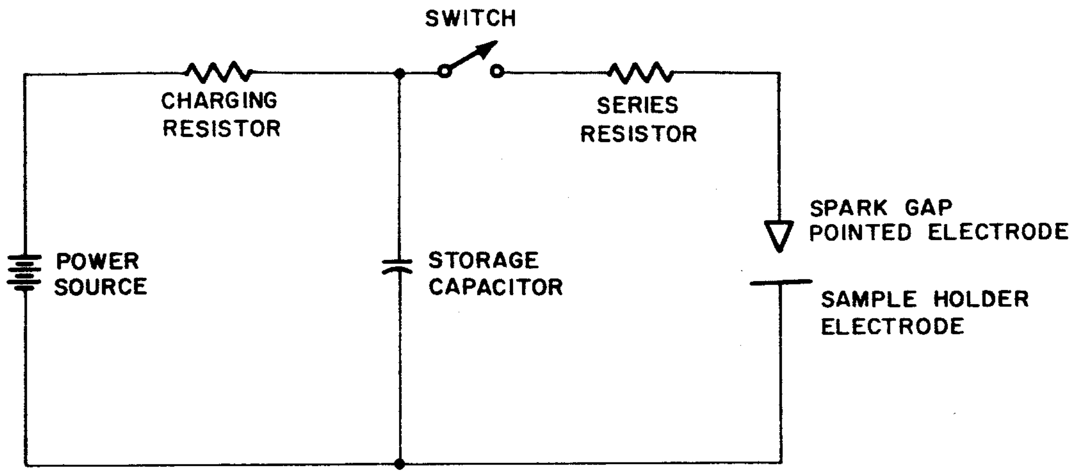


FIG. 6. FREQUENTLY USED CIRCUIT TO DETERMINE MINIMUM ENERGY FOR EXPLOSIVES

Table 6

MINIMUM ENERGIES FOR IGNITION OF VARIOUS EXPLOSIVE MATERIALS

Material	Comments on Preparation or Testing	Minimum Energy on Firing Capacitor (Ergs)
Copper Acetylide		20
Lead Styphnate	Basic Preparation (chemically)	30
	Normal; Energy measurement depends upon experimenter	140 to 9000
	Prepared in humidity less than 0.1%	3.8
	Prepared in humidity less than 1.8%	112.5
	Graphite added in amount of 1%	0.6
Lead Azide	Crystalline	400 to 18,000
	Dextrinated	70,000 to 280,000
Lead Dinitro-resorcinate		500
Mercury Fulminate		
	Unconfined	800,000
	Confined	200,000 to 250,000
	Series Resistance of 5,000 ohms	86,000
	Series Resistance of 250,000 to 750,000 ohms	37,500
Tetracene		100,000 to 280,000

contained in the space vehicle, and this is quite true. This information points out that it may be misleading to determine ignition energy under one condition of circuit constants for static safety and then to apply these across the board for all values of capacitance, resistance and voltage. This is brought out in Figure 7.

In looking at the values of resistance for which energy requirements are relatively low, it appears that some are near the leakage values of certain components. A resistance of 750,000 ohms differs not greatly from that of a contaminated insulator; and by the rules set down previously, this might be just the value of series resistance for firing from a small capacitor with minimum applied energy.

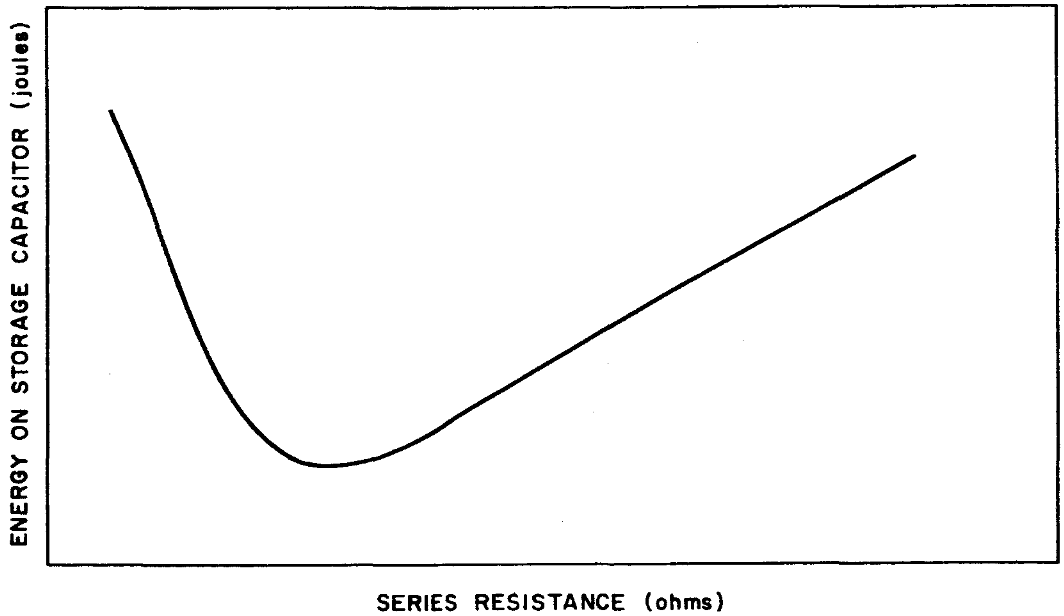


FIG. 7 RESPONSE OF TYPICAL EXPLOSIVE MATERIAL TO SPARK GAP WITH SERIES RESISTANCE

Caution must be observed in the application of information on static electricity to initiator circuits. At this time there is inadequate and incomplete information for any real assessment of the static hazard. More theoretical and experimental work is needed.

3.4.3 Thermocouple Effects

Dissimilar metal or semiconductor junctions are relatively common in most modern structures. Such junctions form a thermocouple and if the junction is then heated or cooled a potential can be produced. Preliminary consideration of this effect with respect to spacecraft leads to the conclusion that potentials and currents will be small. The largest current to be expected is of the order of a few milliamperes. These levels are probably too low to be of any serious concern to EED's of the type to be used in spacecraft with the one somewhat remote exception of the case where the current is fed directly to the bridgewire and is continued for an extended period of time. This once again raises the possibility of cook-off or degradation of the explosives.

3.4.4 Ionizing Radiation

Ionizing radiation from whatever source - solar flares, Van Allen Belt, internal nuclear apparatus, or other - may have indirect electrical effects on components of the initiator or the initiator circuit. Usually these effects are small, and except in unusual circuit applications, where resistance and frequency are critical, may be neglected. An observation has been noted of a 20% change in the magnetic saturation flux density of Permalloy, that looks as if it could be troublesome to firing circuits that employ transformers, relays, or filtering devices using this or other similar magnetic materials. Changes also occurred in the shape of the hysteresis loop, the loop becoming thinner with indications of less loss per cycle. Exposure density was 10^{10} particles per square centimeter to bring about these changes.

Other radiation effects on initiator circuit performance include conductivity of insulators and conductors. These effects appear to be small for the radiation levels thought to exist. Little seems to be known, in this area, and experimental evaluation of radiation effects is even more limited.

This subject is treated at greater length in Section 3.5, "Particle Environments in Space."

3.4.5 Lightning

Exterior Effects

Because of the tremendous transient currents produced, lightning discharges must be considered as a potential hazard to explosive devices. There is no universal agreement as to the electrical properties of lightning discharges, but a few general statements can be made. Cloud-to ground discharges are apparently less intense on the average than cloud-to-cloud discharges. Cloud-to-ground discharges can be characterized as having average currents on the order of 20 to 50 kiloamperes with maximum values approaching 150 kiloamperes. The total charge available could approach 16 coulombs. Cloud-to-cloud discharges have similar average and maximum ampere ratings but the total charge can approach 200 coulombs. With such enormous energies available it is apparent that lightning discharges in the vicinity of spacecraft must be considered as a potential hazard to explosives.

A considerable amount of data has been accumulated on the effects of lightning on conventional aircraft. The frequency of being struck by lightning is one strike per 2500 miles for propeller driven aircraft, one strike per 3800 miles for jet-propeller driven aircraft and one strike per 10,400 hours for jet aircraft. The low incidence for jet aircraft is due in part to the high altitudes flown by such aircraft where there are only relatively infrequent lightning discharges. In fact the main lightning discharge problem occurs below 24,000 feet on earth. Lightning on other planets has not yet been studied.

Direct strikes on conventional aircraft have resulted in damage ranging from very light to extensive, depending upon the nature of the target. Usually troubles occur in the metal skins, in the metal-plastic or pure plastic structures,or across poor bonds. In short, the trouble occurs whenever the target material has a reasonably large resistance. Natural lightning strikes have caused holes up to 4 inches in diameter on aircraft skins.

It would seem that even with a direct lightning strike explosives with a thick low-resistance casing would be reasonably safe insofar as the initiator itself is concerned. If thin-walled charge containers are used, such as is used on some of the lead and plastic enclosed linear charges, intermediate explosive such as PETN and RDX could probably be detonated.

A direct strike on the electrical leads or associated circuits of an electrically initiated explosive component could be disastrous. Usual electrical shielding methods would be of little help in defeating the currents that result from a direct strike.

The electrical fields produced by a near miss create a problem similar to the RF hazard, and the types of circuits required to minimize the RF hazard will minimize pickup from a near miss field also. The fields from the lightning discharge are transient, although of much greater initial intensity.

With regard to launchings from Earth, the choice of the exact time is somewhat flexible, and because high altitudes are reached rapidly it would seem that the best defense against lightning is to provide adequate protection while on the ground and not to launch during a thunderstorm. The choice of time and place for the return to earth, of course, is much less flexible, and thunderstorms might prove unavoidable on occasion.

Interior Effects

The effects of lightning discharges on the internal portions of simulated rockets and spacecraft structures has been under study by the Lightning and Transients Research Institute[25]. The work of this group included experiments on a cylindrical vessel of aluminum with a wall thickness of 90 mils. This vessel was subject to energy from the discharge of a generator capable of delivering current surges up to about 100,000 amperes. Surges were delivered either through a sending probe that consisted of a straight grounded conductor parallel to the cylindrical axis of the vessel or by a direct discharge through the cylinder from the top surface. Fields inside of the cylindrical vessel were checked with a loop 10 cm in diameter connected to a Tektronix 321 oscilloscope.

An analysis indicates that fields inside the cylinder are the result of either a conductive voltage drop on the inside of the wall or a combination of the mutual inductance between the wall and some center conductor, and the self-inductance of the wall. If the wall and the conductor are coaxial, the voltage appearing between the bottom of the center conductor and the wall is given by $(L-m)(di/dt)+iR$.

where

L = inductance of the shield in henrys

M = mutual inductance between inner conductor and shield in henrys

i = current flowing in the shield in amperes

R = resistance in the shield in ohms

t = time in seconds

In this ideal case $L = M$ because of perfect mutual coupling between the cylinder and the conductor. It was demonstrated that in the practical case there are discontinuities in the wall due to joints, holes and access ports that cause L and M to be unequal.

Practical experiments were run to show that there were indeed fields within the cylinder. With a top cover bolted onto the cylinder and with a butt joint between the two halves of the cylinder, a number of measurements were made with the instrumentation described earlier. Passing a current of 70,000 amperes through sending probe or through the cylinder itself resulted in the delivery of a voltage pulse inside the cylinder that showed ringing, decaying exponentially with time. Maximum voltages of 100 millivolts appeared on the loop, at the point where the cover was joined to the wall of the cylinder. This was true whether the current was applied through the sending probe or through the wall of the cylinder. The amplitude of this signal could be reduced by improving the contact of the joint between the cylinder wall and the cover. Tighter bolting resulted in a substantial reduction of the signal and "Heli-arc" welding of the top reduced the signal by orders of magnitude. Near the bottom of the welded tank, the loop signals were on the order of 260 millivolts. The bottom of the tank was open in this case.

Increasing the distance between the sending probe and the tank resulted in a drop in induced voltage, according to a distance function somewhere between linear and square law.

Penetration of energy near a 2-inch diameter hole in the wall of the cylinder resulted in a maximum induced voltage of about 100 millivolts.

All of these investigations on the penetration of energy through the wall are interesting but they do not demonstrate whether explosive devices are safe within the confines of space vehicles and rocket cases. With the loop size used and with the instruments described, and because the loop itself probably has a very low impedance, the potentials indicated are probably the open-circuit potentials of the loop. The power delivered to a one-ohm load resistor would be no greater than about 90 milliwatts, and would be a few milliseconds in duration at the most. Under these conditions there is little need to be concerned for the safety of electroexplosive devices or for other cased charges. However, larger pick-up loops or larger openings in the wall could prove hazardous. There was sparking observed on the inside wall of the cylinder near joints in the surface. This could well present a problem if the lead wires of circuits containing EEDs were attached to the inside wall of the cylinder.

The reference cited(25) also discussed problems concerning the effects of lightning discharges on cables. Here the theoretical situation is the same as that discussed for the cylinder. There are two modes of delivering energy into a coaxial cable, that of the current-resistance product that causes a potential drop on the inside of the cable and that of mutual inductance of the center conductor to the shield and the inductance of the shield itself. The action of the shield in the presence of a lightning discharge is illustrated in Figure 8. The resistance of the shield combined with the passage of current through it results in a potential difference within the shielded portion of the circuit. Though the resistance of the shield is small, the discharge currents are of great magnitude and could produce potentials that are

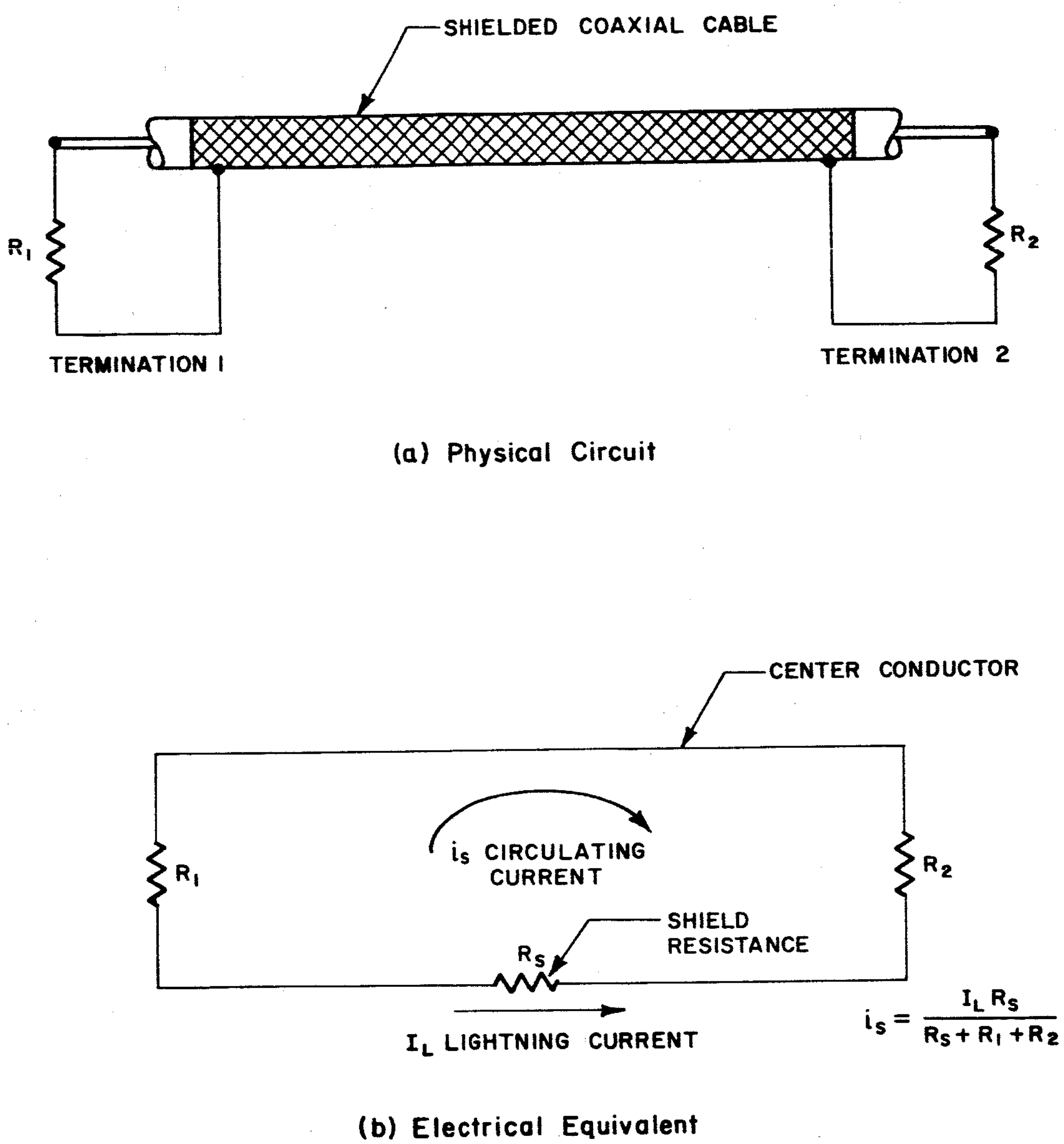


FIG. 8. THE EFFECT OF LIGHTNING STRIKES ON COAXIAL CABLES

high. This was illustrated by firing an exploding bridgewire (EBW) "squib" in the center of a 100-foot length of RG-8/U cable. A current of 100,000 amperes was passed through the shield and both ends of the cable were shorted. Then, with the ends of the cable open, another attempt was made to fire an EBW device, but without success.

Entrance of the energy into cables can be a problem in that the shield can be damaged by very high currents from say, a lightning stroke. Currents in the shield could cause loss of sensitivity even with safe conditions existing otherwise.

High magnitude pressures are also generated by near strikes, but these are generally not large enough to initiate either primary or high explosives.

3.5 Particle Environments in Space

Future explorations to Venus, Mars, the asteroids, etc. will require the space vehicle to endure the environments to be encountered there for times as long as 18 months or more. Hence, factors of minimal consequence in low earth orbiting may be expected to take on new significance. Of those environments comprising particle matter ranging in size from that of the electron to the boulderlike asteroid, the greatest potential hazard arises from proton and electron bombardment produced by solar flares, and from meteoritic collisions. A review of these environments and their effect upon space vehicle electroexplosive systems follows.

3.5.1 Proton-Electron Flux

The potential danger from solar particulate radiation is best estimated by assuming the solar flare to be typical of the worst case. During these occurrences it is not unusual for the ambient radiation level to be increased by five or more orders of magnitude. Data indicating the intensity of solar flare activity is given in Table 7. Characteristics of protons and electrons found in the Van Allen belt are also given.

Preliminary design considerations to be drawn from these data are that a nominal proton/electron flux on the order of 10^8 particles/cm^2 could be reasonably assumed. This flux will comprise particles of about 40 Mev average energy and may be assumed to exist anywhere within 15 earth radii of the earth.

The degree or range to which an electron or proton will penetrate matter is ordinarily given in terms of mass per unit square. For any arbitrary surface area a larger mass, hence increased thickness, is required to stop electrons and protons of greater energy.

A typical plot of this relationship for electrons in aluminum, silicon and silicon dioxide is shown in Figure 9. Ranges for other materials would be represented by lines parallel to the line in the figure. On the basis

Table 7

PROTON AND ELECTRON ACTIVITY NEAR THE EARTH

Solar Flares	Particle Energy* (ev)	Average Flux (particles/cm^2-sec)	Power Density (watt/M^2)
Maximum Recorded - 1960 (Protons)	$>30 \times 10^6$	2×10^4	9.6×10^{-4}
Maximum Recorded - 1964 (Protons)	$>6 \times 10^6$	9×10^7 to 50×10^7	>4.8
Typical Flare: (pre-1964 data) Protons	4×10^4	7×10^8	4.5×10^{-2}
Electrons	10^5	5×10^6	8.0×10^{-4}
Typical Flare (later data)			
Low Energy Protons	120×10^3 to 4.5×10^6	10^8	1.9×10^{-2} to 0.7
Electrons	110×10^3 to 1.6×10^6	10^8	1.7×10^{-2} to 0.2
High Energy Protons	30×10^6 to 700×10^6	4×10^4	1.9×10^{-3} to 4.5×10^{-2}
Electrons	1.6×10^6 to 5×10^6	10^5	2.5×10^{-4} to 8.0×10^{-4}
Van Allen Belt			
Protons	40×10^6	13×10^4	7.4×10^{-3}
Electrons	$>2 \times 10^4$ to $>6 \times 10^5$	6×10^9 to 6×10^7	0.19 to 4.5×10^{-2}

*Typical values of mode of particles; (>) symbol indicates top level of detector.

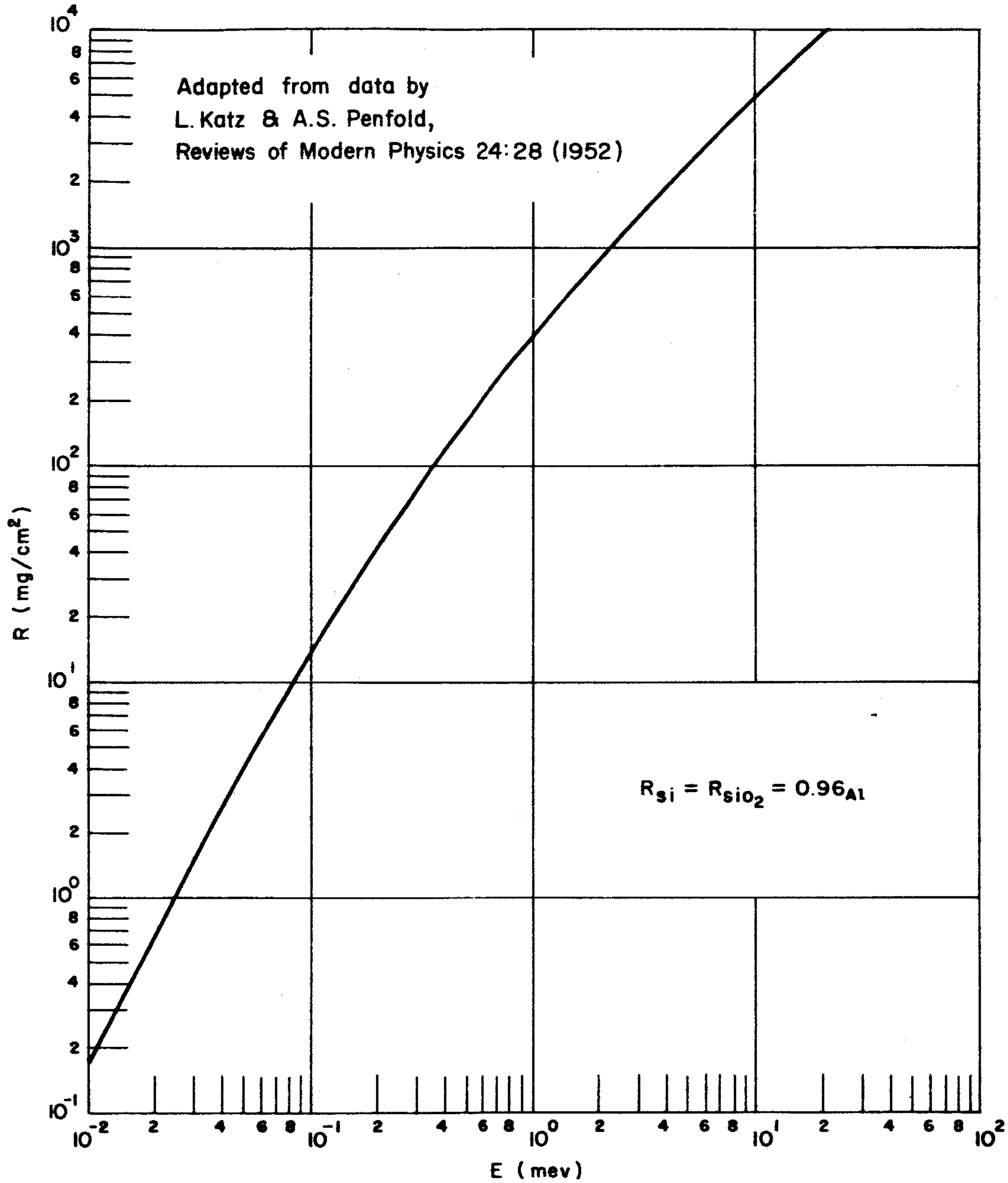


FIG. 9 RANGE – ENERGY RELATIONSHIP FOR ELECTRONS IN ALUMINUM

of these data, a 10-mil thickness of aluminum foil will stop Van Allen electrons. However, to stop electrons on the order of 5 Mev, such as occur during an intense solar flare, 400 mils of aluminum shielding would be required.

The stopping distance (range) in matter for protons is described exactly the same as for electrons. In Figure 10 the range for several materials is shown in relation to incident proton energy.

As stated earlier 400 mils of aluminum will stop solar flare electrons of 5 Mev, this amount of shielding would also stop solar flare protons of 50 Mev. However, intense solar flares produce protons on the order of 100 Mev; 3.94 inches of aluminum shielding would be required to completely stop protons of this energy.

Some EEDs are cased in 1/16 inch aluminum. This would stop protons of about 20 Mev and electrons of 1.5 Mev.

Figure 10 shows that hydrogen is about 250% more effective per unit mass for stopping protons of a given energy. If some means of using hydrogen as a shielding material could be found (such as a balloon-like container) greater savings in shielding weight would result.

A sample of the data used in Figure 10 is presented in Table 8.

3.5.2 Chemical Effects of Particle Bombardment in Space

The sun, which is the dominant radiating body in the solar system, emits particles as well as other forms of radiation. The most energetic of these solar particles are protons in the Mev range. There are lower energy particles including protons, electrons and some helium nuclei whose energies lie in the 100 electron volt range. The density of these particles starts at about 3×10^7 atoms/cc (3×10^7 protons plus 3×10^7 electrons) in the vicinity of the sum and decreases continusouly to at least as far as the orbit of the Earth where it is only about 10 electrons/cc. Solar proton density in the vicinity of the Earth is very low, 5 to 20 protons/cc. These are quiet-Sun values.

Table 8

RANGE-ENERGY RELATIONSHIP*

E (Mev)	Be	Al	Cu	Ag	Pb	Air	H_2O	Stainless† Steel
2.5	1.30×10^{-2}	1.64×10^{-2}	2.34×10^{-2}	2.95×10^{-2}	4.33×10^{-2}	1.28×10^{-2}	1.09×10^{-2}	2.40×10^{-2}
5.0	4.31×10^{-2}	5.15×10^{-2}	7.00×10^{-2}	8.67×10^{-2}	1.20×10^{-2}	4.19×10^{-2}	3.60×10^{-2}	6.50×10^{-2}
10	1.48×10^{-1}	1.7×10^{-1}	2.20×10^{-1}	2.65×10^{-1}	3.57×10^{-1}	1.42×10^{-1}	1.23×10^{-1}	3.60×10^{-1}
25	7.64×10^{-1}	8.50×10^{-1}	1.04×10^{0}	1.22×10^{0}	1.59×10^{0}	7.39×10^{-1}	5.94×10^{-1}	1.00×10^{0}
50	2.71×10^{0}	2.92×10^{0}	3.52×10^{0}	4.05×10^{0}	5.11×10^{0}	2.55×10^{0}	2.24×10^{0}	3.1×10^{0}
100	9.44×10^{0}	9.97×10^{0}	1.18×10^{1}	1.34×10^{1}	1.65×10^{1}	8.82×10^{1}	7.77×10^{1}	1.0×10^{1}
250	4.65×10^{1}	4.84×10^{1}	5.65×10^{1}	6.32×10^{1}	7.62×10^{1}	4.32×10^{1}	3.82×10^{1}	5.4×10^{1}
500	1.44×10^{2}	1.48×10^{2}	1.71×10^{2}	1.91×10^{2}	2.27×10^{2}	1.33×10^{2}	1.18×10^{2}	
1000	4.00×10^{2}	4.09×10^{2}	4.69×10^{2}	5.20×10^{2}	6.16×10^{2}	3.69×10^{2}	3.28×10^{2}	

Range given in gm/cm^2

*American Institute of Physics Handbook 2nd ed. 1963 Mc Graw-Hill, N.Y.

†Data on Stainless Steel added; based on graph, by Evans of M.I.T., found in: Principles for the Calculation of Radiation Dose Rates in Space Vehicles: NASA Report 63270-05-01.

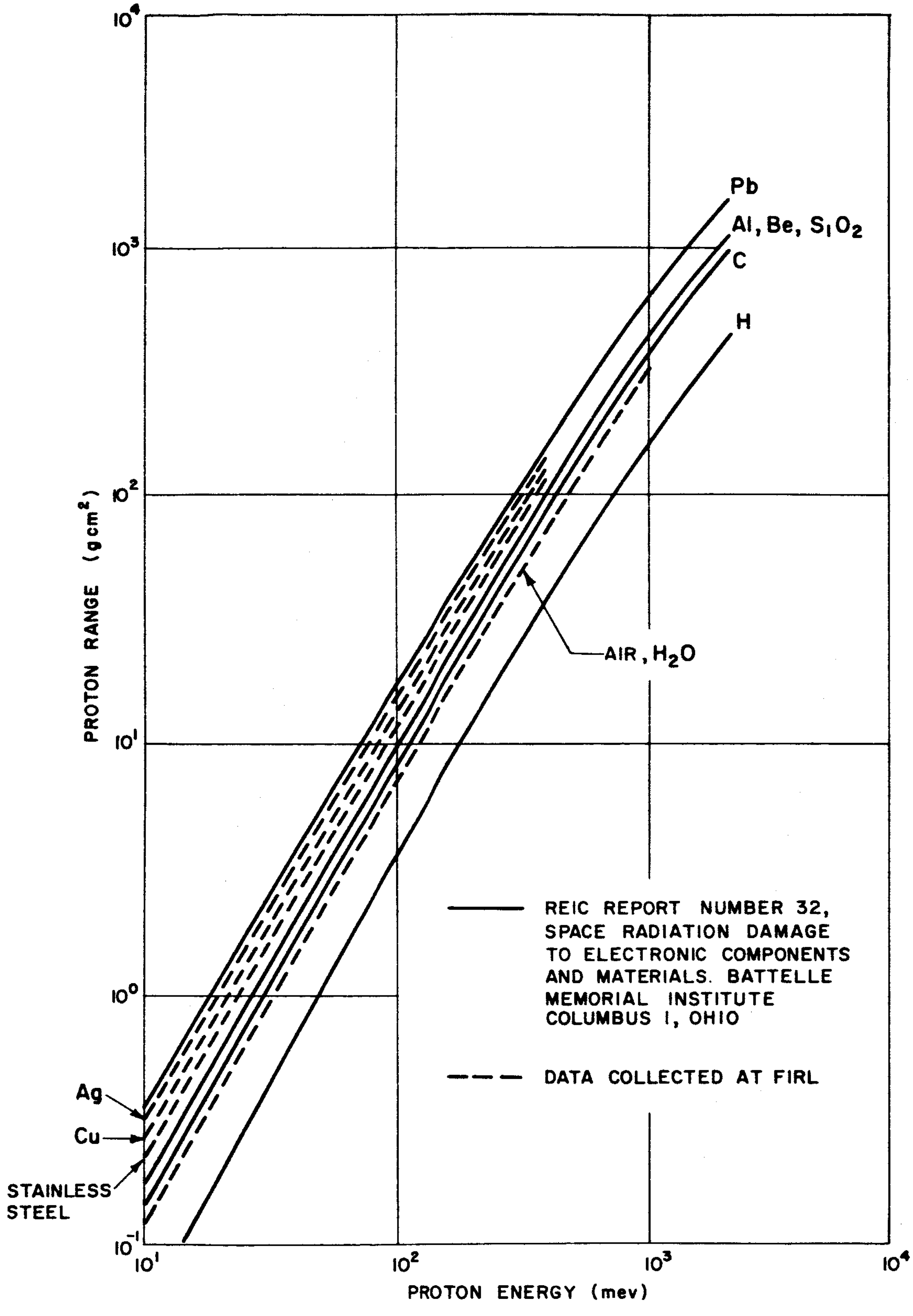


FIG. 10. RANGE OF PENETRATION AS A FUNCTION OF PROTON ENERGIES FOR SOME MATERIALS

In passing through any medium, each high energy particle loses energy by interacting with electrons and nuclei of the medium.

The basic mechanisms of energy interchange are:

(1) Ionization - in which an orbital electron is removed from its parent nucleus giving rise to a free electron and a positively charged (ionized) atom or molecule.

(2) Excitation - in which an electron is raised to a high energy level but remains bound to its parent nucleus. In this case the atom or molecule remains neutral.

(3) Displacement of a nucleus with or without its attendant electrons.

(4) Capture by an atomic nucleus and transformation of the nuclear structure.

(5) Scattering of the incident particle or photon and emission of secondary radiation.

These basic processes may given rise to secondary charges. Thus an unstable nucleus may be formed in process (4) which on disintegration emits a further high energy particle capable of inciting further ionization, excitation or nuclear displacement. The electron emitted during ionization may also have sufficient energy to cause secondary ionization and excitation in neighboring atoms until it loses most of its energy and is then captured by the same or another atomic nucleus. Much of the energy absorbed within the specimen will be degraded into thermal vibrations and eventually appear in the form of heat.

When we consider the effects on various materials it appears that we can make some generalizations with regard to metals, covalent and ionic structures, and molecular structures.

The only effect of exposure of metals to high energy radiation comes from the displacement of atoms giving rise to interstitial atoms and lattice vacancies. Conduction electrons already present in the metal react with any ions formed and prevent radiation-induced ionization or excitation from causing any permanent change in properties. Changes in atomic arrangements will lead to changes in electrical resistance, elastic modulus, creep, yield strength, stored energy due to the distorted lattice, hardness and ductility. For many metals these effects anneal out at room temperature. Approximately 10^{18} protons/cm^2 are required to change the brittle-ductile transition temperature in mild steel only 19°C. A similar level is required to change electrical resistivity; for example, 10^{16} protons/cm^2 are needed to effect a change of 0.5% in tungsten. The quiet emission from the sun will have virtually no effect and even the maximum integrated flux of Van Allen protons, which is about 10^{12} protons/cm^2-yr in the heart of the belt, indicates that the level of penetrating and bombardment flux encountered in space is too low to cause significant damage to metals.

In covalent and ionic structures the same considerations of conservation of momentum apply to the amount of energy which can be transferred to an atom by collision, and this therefore sets a lower limit to the energy of each incident particle below which no atomic displacements are possible, and only ionization and excitation effects can be obtained. An alternative mechanism has been suggested, however, that applies to ionic lattices. If an atom suffers multiple ionization, for example by low energy x-rays incapable of causing displacement directly, a situation can arise whereby a negative ion surrounded by positive ions may be positively charged and then be displaced because of electrostatic forces acting on it. This theory would account for displacements of the atomic nuclei in crystalline lattices at radiation energies too low to allow displacement by collision. The changes observed include elastic constants, energy content, and thermal conductivity. These properties are often recoverable by subsequent thermal treatment, although not all of the properties may become annealed out simultaneously. Materials in this grouping include diamond, graphite, quartz, germanium, silicon, zircon, boron nitride and silicon carbide.

It is easy to see that silicon and germanium are sensitive to bombardment by high energy particles since their electrical properties are controlled by doping with concentrations of 1 in 10^7 atoms and the carrier recombination properties are usually controlled by residual impurity atoms or defects in similar concentrations.

In molecular structure where atoms are bound together by shared electrons to form molecules, excitation and ionization are by far the most important effects of exposure to high energy radiation. Radiations which do not already consist of fast charged particles give rise to charged particles and these cause the chemical changes observed. The high energy particles act by causing a nearly random ionization and excitation along their track, but the randomness does not persist, and, even before chemical bonds are broken, the effect of the energy tends to concentrate at certain positions. There is a definite tendency for weaker bonds to be broken by radiation. The removal of an electron, or even its excitation to a higher level, may render such molecules unstable and decomposition ensumes. The active fragments produced can react with each other or with neutral molecules to yield molecules which are chemically very different or structures which may themselves be unstable to lead to further reactions. In this way one ionization or excitation can lead to a number of reactions, and in suitable systems the number of molecules modified per ion pair produced by radiation may be very high. Although the damage to organic compounds is usually a function of energy there are cases where energy can be absorbed and gradually dissipated by a resonant structure like the benzene ring without breaking chemical bonds. As an example, it takes 1400 to 1600 ev of energy absorbed in polystyrene to break chemical bonds (based on total energy required for one crosslink) as compared with 45 to 60 ev in polyethylene. These two polymers have the same vinyl type backbone structure except that in polystyrene a hydrogen atom is replaced by a phenyl group.

In general it appears that for interplanetary voyages (elapsed time of 10^8 seconds) we need not be concerned with the chemical effects upon metals that may be used in initiator designs but that we should be careful that any insulators or organic materials which may be exposed are of types that will not be seriously impaired.

3.5.3 Space Debris

It is recognized that various forms of matter in interplanetary space must be considered as a hazard to space navigation. This matter can be classified as asteroids, meteoroids and dust. Due to limited amount of data no firm conclusions as to the effect of this matter on spacecraft can be drawn at the present time.

What information is available indicates that the asteroid belt will pose the least problem for deep space flight. The belt lies between the orbits of Mars and Jupiter; it occupies approximately 10^{16} cubic miles of space and contains less than 10^{6} asteroids of all sizes. If the asteroids were evenly distributed throughout the belt, each would be separated by 1000 miles from its nearest neighbor in any direction. The total mass of all asteroids combined is about one one-thousandth of the earth's mass (26) and would occupy a sphere 625 miles in diameter.

All of the 1500 catalogued asteroids rotate about the sun in the same direction as the earth. Therefore, a spacecraft in a transfer orbit should easily be able to pass through the belt without significant danger. This presumes that the craft has the ability to move laterally (e.g. Gemini) and is initially matched to the average velocity of the asteroid belt.

Hazards due to chance collision with meteroids and micrometeoroids will increase as we probe toward the asteroid belt. Deformations and punctures are the obvious forms of damage. In addition, however, we must be concerned with the likelihood of heat and sparks, and with the electrical charge transfer that might occur during impact.

3.5.4 Meteoroids

On any clear night a casual observer could probably see about 5 to 7 meteors per hour, coming out of the sky from random directions. These are sporadic meteors. There are also meteor swarms, large numbers of meteoroids traveling together about the sun in the same orbit. When they intersect the earth's orbit a meteor shower occurs.

The known meteor swarms travel about the sun in highly elliptical orbits, and many are associated with the orbits of former comets that have been broken up by the gravitational pull of Jupiter.

The Russians have reported that both their Mars 1 and Mars 2 probes were damaged by previously unknown meteor swarms. This is indicative of the hazard that meteors pose to spacecraft, and demonstrates the need for further study to determine the existence of presently unknown swarms, and to compute their orbits. It is strongly recommended that the trajectories of spacecraft be chosen so as to avoid the likelihood of an encounter with a meteor swarm; the requirement for a consultant astronomer is inescapable, in planning a mission.

The problem of protecting an EED from meteoroid impact becomes evident when the range of meteoroid velocities is considered. A meteoroid in the vicinity of the earth cannot travel less than 9 km/sec or it would fall into the sun. The maximum permissible velocity (except for a non-orbiting stray) is 42 km/sec which is escape velocity of the solar system. The maximum velocity of impact with the earth is 72 km/sec. This is for a meteoroid in a retrograde orbit traveling at 42 km/sec colliding with the earth whose orbital velocity is 29.8 km/sec. If a missile is traveling at the velocity of escape from the earth, 11 km/sec, the maximum velocity of collision would then be 83 km/sec.

Even though micrometeoric dust can have the same velocity as meteoroids it has been found that dust erosion is quite small. While not much data is presently available, a measured flux of 10^{-6} micro-meteoroids per cm^2/sec near the earth is found to result in no more than 10^{-5} cm of the surface area destroyed per year.

3.5.5 Sputtering (Corpuscular Radiation Damage)

Sputtering is the removal of atoms and molecules from a surface bombarded by high velocity particles (solar corpuscular radiation) or individual molecules of highly attenuated gas.

As in the case of proton radiation, the worst case of sputtering will occur during an intense solar flare. It is concluded that sputtering by corpuscular radiation will remove between 0.9 and 6 microns of silver per year. Since micrometeoroids erode 0.01 to 0.1 micron per year, sputtering removes within an order of magnitude as much material as micrometeoroid erosion. It is thought that 4 microns of aluminum at the most could be removed in one year by sputtering due to solar corpuscular radiation.

The sputtering damage to aluminum at an altitude of 700 km is on the order of that caused by micrometeoroids, which should be expected to remove 0.0001 microns in about 6 hours; silver, however, is not damaged at all, which indicates the varied effects of sputtering on different materials. Sputtering by the atmosphere at altitudes above 700 km is not considered to be serious or dangerous, except for the possibility of small surface change or slight damage to optical surfaces.

3.5.6 Accumulated Charge

The kinetic electron volt potential developed when a particle strikes a target has been related empirically to accumulated charge, which is given by

$$Q_C = \frac{KE_k V}{A}$$

where Q_C = charge collected
E_k = kinetic energy of particle
K = constant for target material
A = atomic weight of particle
V = velocity of particle

The relationship of K has not been determined as yet but it has been shown to be much larger for some materials such as tantalum and tungsten than it is for others such as copper, beryllium, copper alloy, and iridium. This phenomenon is worth added attention in view of the fact that many EEDs are very sensitive to relatively small charges.

3.5.7 Conclusions

We conclude from present available data that the only catastrophic hazard due to space debris is the possibility of meteoroid collision. More data are needed, especially concerning still undiscovered meteor swarms in interplanetary space. Every effort should be made to avoid an encounter with any meteor swarm.

Sputtering and micrometeoroid erosion present no significant hazard to EEDs, though some metallurgical surface effects might result.

More work is needed in the field of hypervelocity impact (more than 22 km/sec). Within the limits of presently available data we can postulate that a 1/16-inch (62.5-mil) case for an EED should suffice to protect against puncture by micrometeoroids, erosion by dust, and sputtering. The danger due to impact with golf-ball size meteoroids is obvious; fortunately the probability of this occurrence is correspondingly lower. Additional information and investigation is necessary before the full extent of this hazard source may be defined.

3.6 Shock and Vibration

3.6.1 General Considerations

The EEDs under consideration will be subject to a severe shock and vibration environment in combination with a vacuum and extremes of temperature. Such an EED has as part of its structure an electrical connector body with two or more pins extending through a metallic bulkhead; the pins are hermetically sealed with glass, ceramic, or epoxy. The mating electrical connectors are keyed to fix the orientation of the pins. The connector of the EED is externally threaded to accept the rotating sleeve of the mating cable connector, in order to provide a metal to metal seal which will remain firm under all conditions. On the other side of the metallic bulkhead containing the hermetic seals, the extensions of the connector pins will be used to support the one or more bridgewires of the EED. Generally the connector pins are potted from the bulkhead up to the bridgewire, to prevent movement of the pins and breakage of the bridgewire. Packed in proximity to the bridgewire are one or more explosives; these are contained in a metal shell which must be strong enough to contain and direct the explosive force. The complete device itself is contained or attached in some manner to the object to which the explosive force must be supplied.

Consider now the possible failure modes of the EED under study, beginning at the cable connector to which it is attached.

1. Loss of electrical continuity at the junction of the EED connector pins and the mating connector.

2. Mechanical failure of mating threads of the EED and cable connector, also failure at the stress concentration point at the transition between these threads and the connector wall.

3. Transfer of the connector cable movement through the EED pins, breaking the hermetic seal.

4. Failure of the hermetic seal, by direct effect of vibrations.

5. Breakage of the bridgewire welds by shifting of connector pins in potting material.

6. Displacement of the explosive mix at the bridgewire with a subsequent change in functioning time.

7. Reorientation of the explosive under long term vibrations to obtain a heterogeneous mixture with altered characteristics.

8. Failure of the EED shell at the point of attachment to the next device when initiation occurs, causing misdirection of the explosive force. Also, failure of shell at other points of stress concentration intended to withstand the explosive force.

It is evident that, even with good design techniques, thorough testing of the EED will be required. But before the shock and vibration levels to be used in testing the EED are discussed, it is necessary to consider the EED, its associated connecting system, and the mounting method.

It is most important that the electrical cable, particularly the connection to the EED and adjacent portions, be as rigidly mounted as the EED itself. Otherwise, it will be possible for the EED connector, the pins, and the hermetic seals to be subjected to excessive stresses well beyond those normally received when the EED is tested alone in the expected environment.

If there is any doubt about the cable mounting, it should be tested along with the EED in the expected environment. The device to which the EED is attached must of course also be rigidly mounted.

The most severe test for the EED would probably be that for combined low temperature vibration and shock environment when the initiator is expected to function. At this time the metal case enclosing the explosive is subjected to the highest stresses, but is least capable of local yielding to relieve them.

Long-term sustained vibrations which might tend to separate the explosive charge from intimate contact with the bridgewire, or to rearrange the designed distribution of the explosive substances, must be considered.

Standard statistical testing techniques to determine any significant changes in the EED functional response after testing in the environment, should be applied.

Furthermore, the EED must function with a given firing pulse, in a given time, with a specified minimum energy output, after a specified minimum energy input has been applied. This must be accomplished at a specified time, whether before, during, or after exposure to shock and vibration. Accordingly, testing of EEDs should include exposure to the expected environment, using MIL-STD-810A as a minimum, and including any other more severe shock and vibration specifications peculiar to the vehicle mission requirements.

Premature firing of the EED due to application of the expected shock and vibration environment cannot of course be tolerated. Such an occurrence during the environmental testing sequence would demand design changes, or the selection of a different type of EED.

3.6.2 Test Specifications

To aid the prediction of shock and vibration design and testing levels to be used as a guide for EED development, selection, and testing, we compare the following:

1. Generalized test envelopes and procedures from MIL-STD-810A (USAF) 23 June 1964, "Environmental Test Methods for Aerospace and Ground Equipment."

2. Design qualification and "off-limit" tests for the Apollo Standard Initiator (ASI) from NASA MSC, Houston, Texas.

3. Maximum level vibration shock and acoustic noise data obtained from flight testing of launch vehicles utilizing high thrust stages, and component acceptance test levels for these spacecraft.

Under each of the headings below there is a comparison of available data and a concise discussion.

Acceleration

MIL-STD 810A: 18g maximum acceleration specified along three mutually perpendicular axes, for a minimum of one minute test on each axis (calculated by assuming ground launched vehicle, liquid booster, forward acceleration of vehicle, and EED position unknown). 150% x 12g maximum = 18g.

Apollo Standard Initiator specification, operational design criterion: Initiator shall perform satisfactorily after exposures to 20g along each of three mutually perpendicular axes, 120 seconds minimum on each axis.

No data presently available indicates that acceleration test levels exceeding 20g (ASI specification) are required for manned aerospace vehicles, or vehicles equipped with liquid boosters; use of solid fuel boosters may require an increase in test levels.

Vibration - Sinusoidal

MIL-STD-810A: For a ground launched vehicle, a rigidly mounted EED and a booster with a thrust greater than 500,000 pounds, the curve shown by the solid line in Figure 11 should be used, for 30 minutes total cycling time.

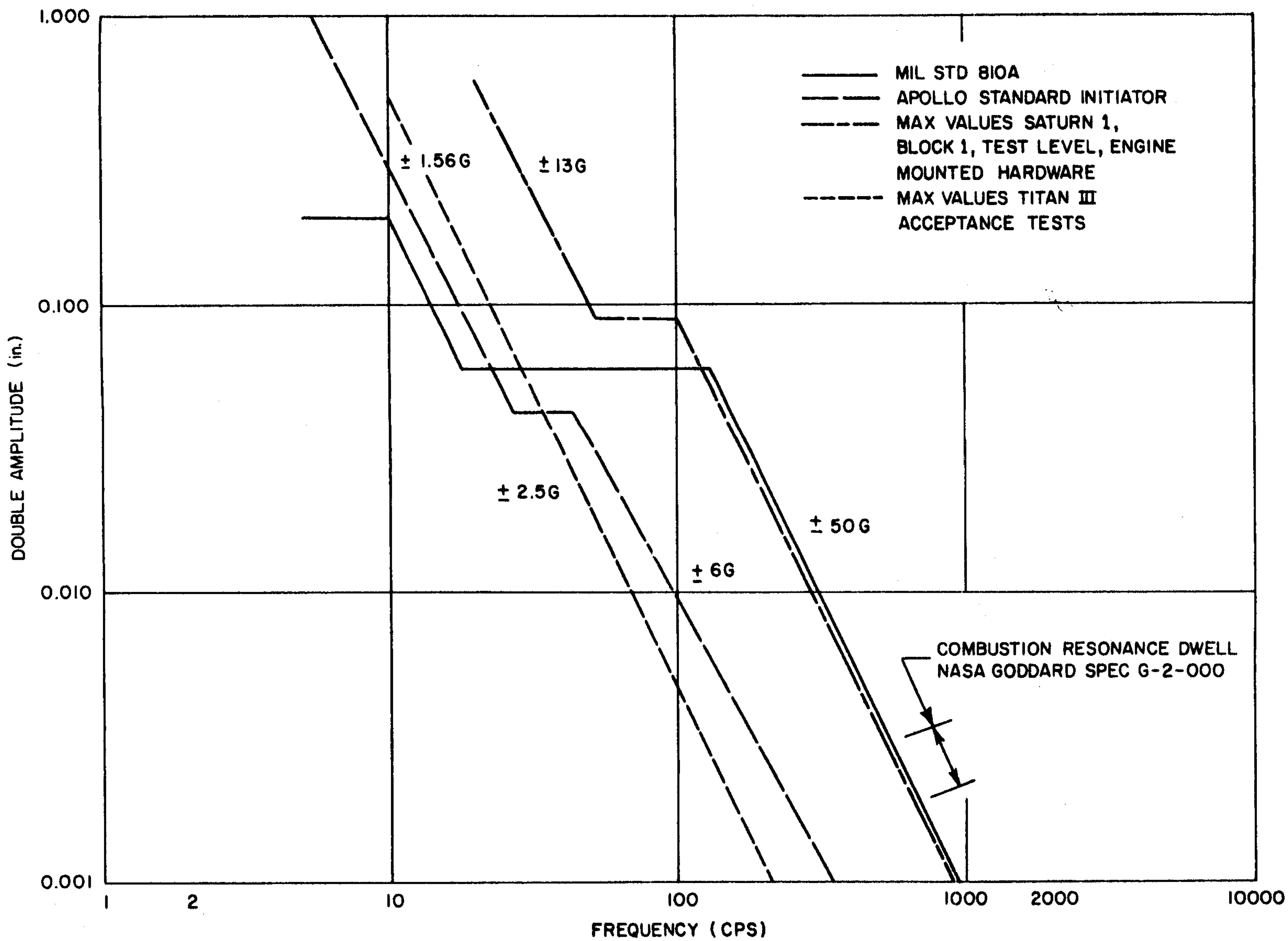


FIG. 11. VIBRATION TEST CURVES
(Sinusoidal)

ASI Specification: As shown by long-dashed line on Figure 11 or according to the following table:

5.0 to 27.5 cps	1.56 g
27.5 to 52.0 cps	0.043 in., double amplitude
52.0 to 500.0 cps	6.0 g

These vibration tests are conducted at ambient temperature, +300°F, and -260°F; one frequency sweep in five minutes per axis; the initiator must perform satisfactorily after tests.

Also shown on Figure 11 are the maximum values obtained on engine-mounted hardware from tests of Saturn 1, Block 1,[27] which are probably the most severe conditions that would be encountered. Maximum values from Titan III component acceptance tests[28] are plotted on the same figure, along with the 500 to 650 cps "combustion resonance dwell", 50 g peak acceleration, as specified by NASA-Goddard Spec. G-2-000.

Vibration - Random

MIL-STD-810A: The random vibration test envelope for a ground-launched vehicle with a thrust of greater than 500,000 is shown in Figure 12. The test duration is 30 minutes per axis.

ASI: Random vibration design criteria levels are listed below:

0.01 g^2/cps at 10 cps with 6 db/octave increase to 0.8 g^2/cps from 100 cps to 400 cps. 0.8 g^2/cps at 400 cps with a 3 db/octave decrease to 0.16 g^2/cps at 2000 cps; 15 minutes test along each axis. These levels are translated into the test curve shown by the dotted line on Figure 12. The initiator must perform satisfactorily during or after exposure to the environment.

Also included are acceptance test levels for Titan III components, and the NASA Goddard Spec. G-2-000. Note that the three acceptance tests shown extend the test frequencies below the 40 cps lower limit of MIL-STD-810A. The ASI program also includes "off-limit" vibration tests, performed at 150,

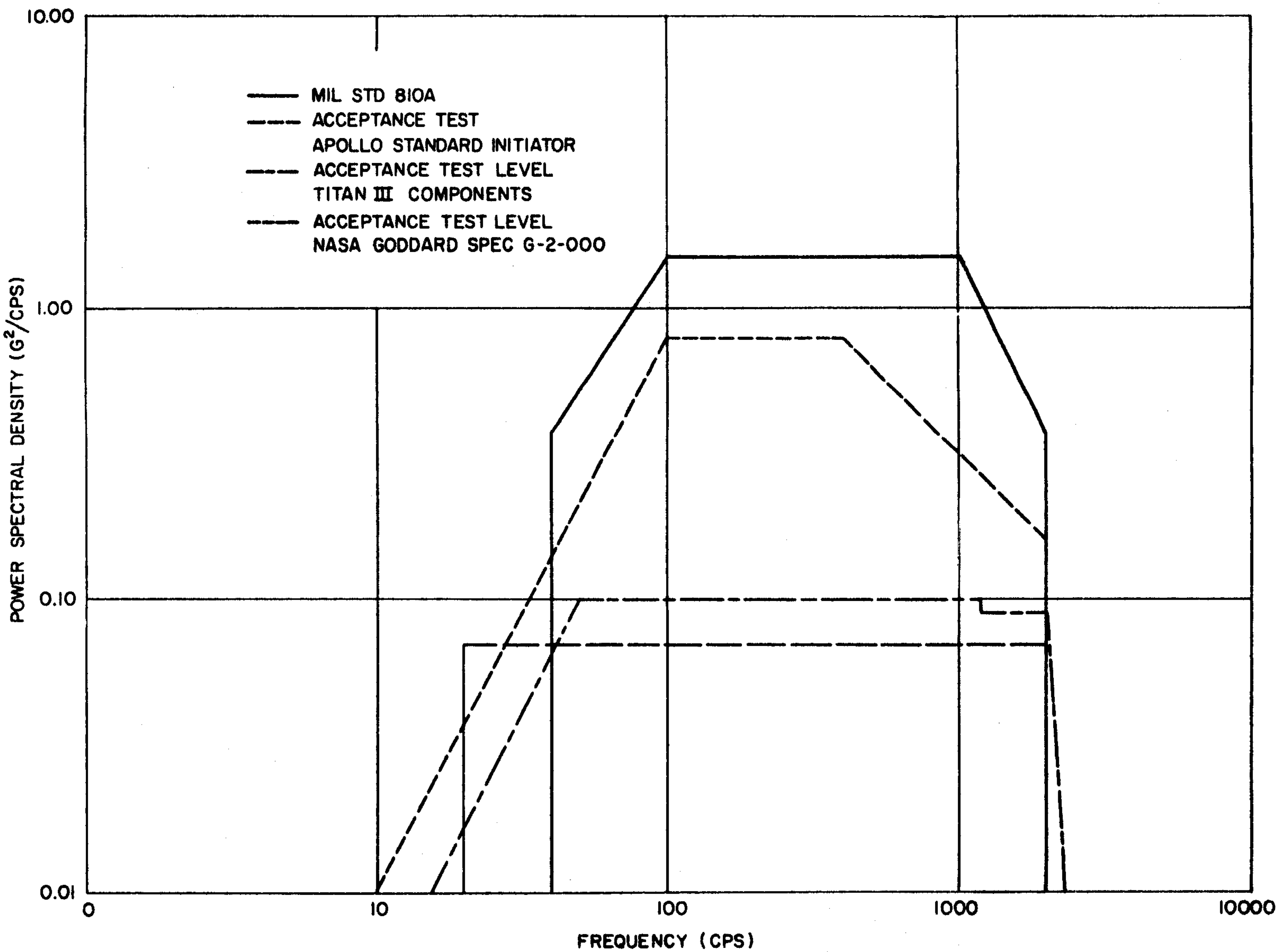


FIG. 12. VIBRATION TEST CURVES
(Random)

175, 200 and 225 per cent of the levels shown. Should test levels higher than the MIL-STD-810A be necessary, failure data from the "off-limit" tests would be a valuable aid for initiator redesign as required.

Acoustical Noise

MIL-STD-810A: The maximum testing level (overall sound pressure level (SPL) of 165 db) from Method 515 is shown on Figure 13.

ASI specification: As shown on Figure 13, the test level shown indicates an overall SPL of 168 db. The test is of 120 second duration with levels as listed below:

cps	db
11.2 to 22.4	154
22.4 to 45	158
45 to 90	162
90 to 180	159
180 to 355	158
355 to 710	155
710 to 1400	150
1400 to 2800	145
2800 to 5600	140
5600 to 11200	135

The average of 600 "on pad" measurements on the SA-2 Saturn[29] is shown by a dashed line on Figure 13. Note that this level is far above the ASI and MIL-STD-810A test levels, and exceeds the highest SPL likely to be encountered by an EED mounted within the vehicle. Since the average EED is very small and of high density, it is not likely to be sensitive to acoustical noise fields[30]. Acoustical noise testing is, however, a valuable supplement to sinusoidal and random vibration tests. It is recommended that some "off-limit" acoustical noise pressure levels, (above 165 db) be used to determine initiator sensitivity and failure modes (if any) to aid designers in perfecting the mechanical construction details of initiators.

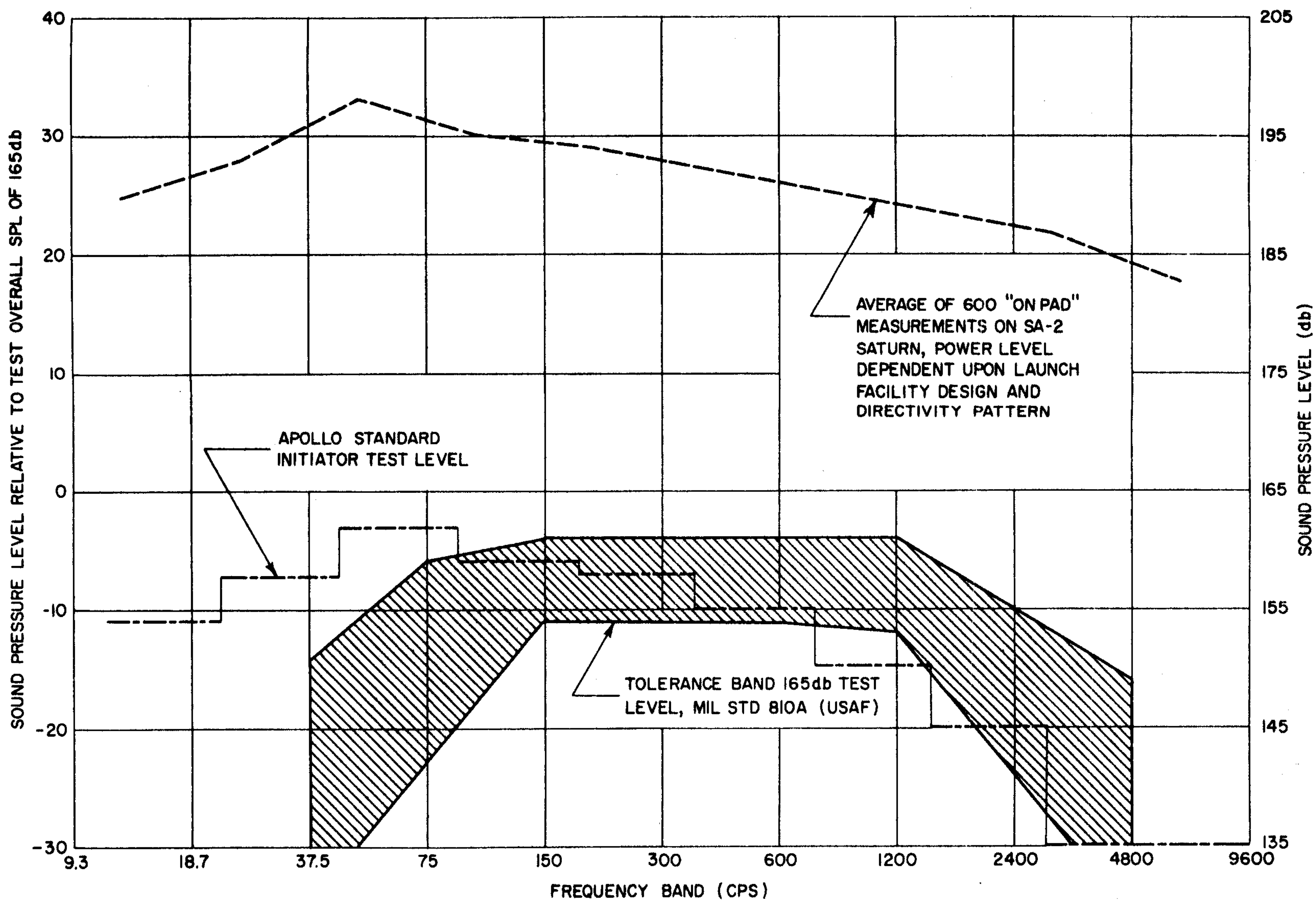


FIG. 13. ACOUSTIC POWER SPECTRA

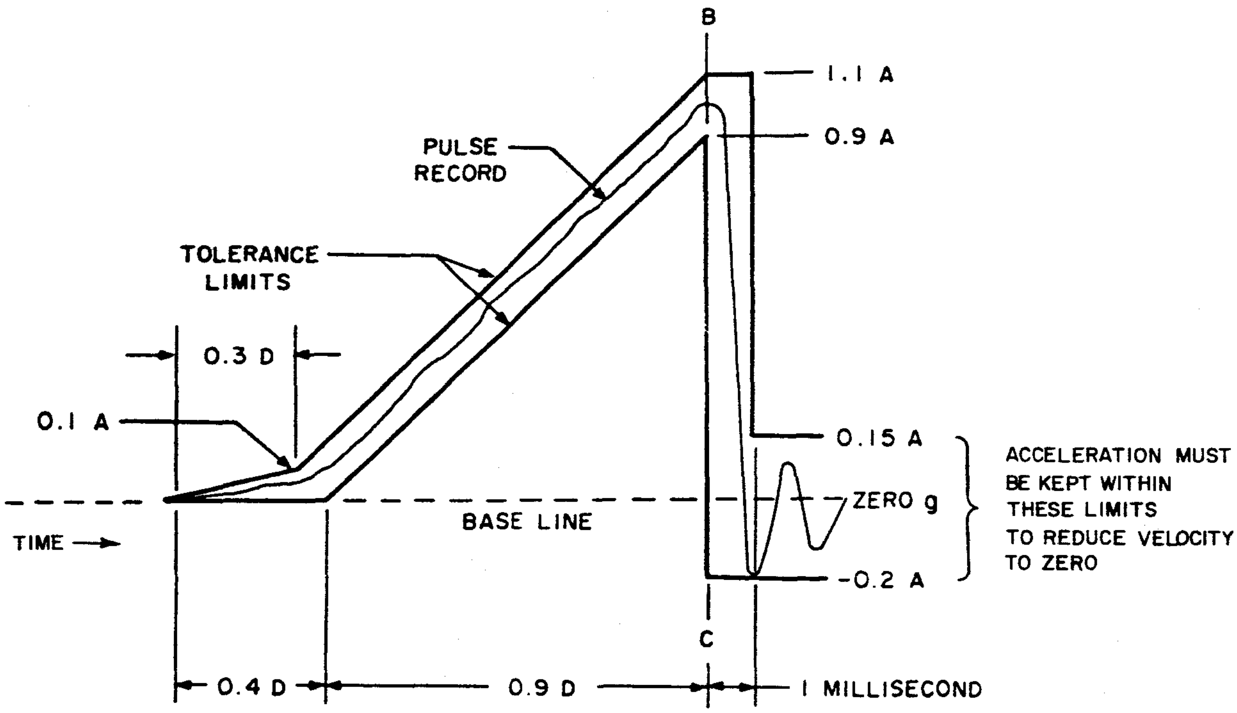
Shock

MIL-STD-810A: Figure 14 is a reproduction of the shock pulse wave form. The high intensity shock test has a peak value of 100 g, with a nominal duration of 6 milliseconds. (Method 516.1).

ASI specifications 30 g for 11 milliseconds is given for an induced environment, but a 100 g test (no time given) is specified for the operational design requirement.

Shock test specification values for typical vehicles are as follows[28]: Discoverer, Midas and Sentry - 40 g maximum with a half-sine pulse of 6 millisecond duration; Titan III - 100 g, non-specified pulse shape of 7 millisecond duration; Block 2 Saturns - 35-150 g depending on location, sawtooth pulse of 5 millisecond duration. The 100 g, 6-millisecond duration sawtooth pulse is recommended for initiator specifications and testing.

Separation of stages, shrouds and payloads when required during missile and satellite spacecraft is usually accomplished by the use of explosive bolts, clamps and frangible joints. Peak accelerations as high as 2500 g with a duration of .5 milliseconds have been recorded[31,32]. A typical acceleration record is shown in Figure 15. Simulation of separation shock by use of a one-half sine wave pulse has been attempted, although a shock machine that will simulate the environment with acceptable realism has not yet been built. A comparison of these two shock spectra is shown in Figure 16. Fortunately the vehicle structure provides significant attenuation of the shock transient, but the amount of attenuation is not always predictable. Failure of brittle components such as glass diodes, resistors and glass seals such as used in EED's are possible depending on component location, although none have yet been recorded. Investigators are pursuing the solution of the problem along three lines: gathering of more complete and descriptive data on the nature of the shock to be expected, translation and correlation of these data by use of laboratory tests and techniques, and reduction of the shock by design of new types of separable joints.



D = NOMINAL DURATION IN MILLISECONDS

A = SPECIFIED PEAK ACCELERATION IN GRAVITIES (g'S)

g = ACCELERATION OF GRAVITY ON EARTH 386 IN./SEC./SEC.

PROCEDURE	TEST	PEAK VALUE A	NOMINAL DURATION D
I	BASIC DESIGN	20 g	10 ms
IV	CRASH SAFETY	40 g	10 ms
V	HIGH INTENSITY	100 g	6 ms

FIG. 14. SAWTOOTH SHOCK PULSE CONFIGURATION

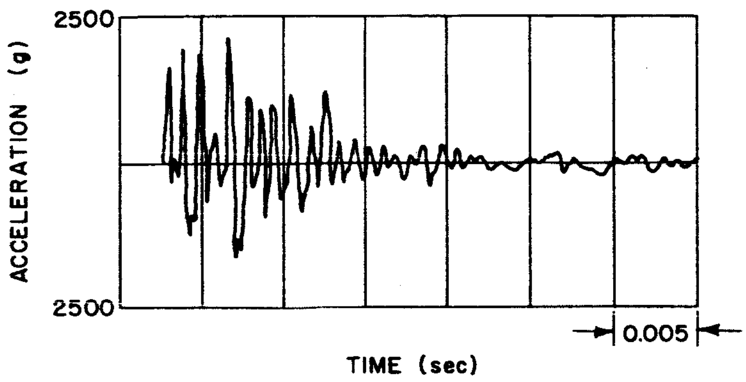


FIG. 15. TYPICAL ACCELERATION RECORD

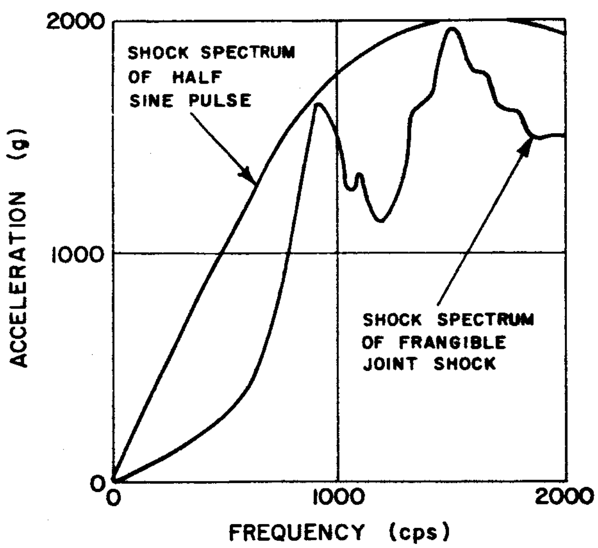


FIG. 16. SHOCK SPECTRA COMPARISON

Testing of EED's under appropriate shock conditions is recommended when methods of producing the necessary test shocks, and correlation data with actual tests on vehicles, are available. Fortunately, research on new types of frangible joints is progressing well, and while data on the magnitude of shock occurring in these designs are not yet available, it probable that acceptable levels will be reached before testing methods for the more severe conditions are perfected.

Not discussed under the above headings are several other associated tests presently performed on the Apollo Standard Initiator, and which are recommended for all EED's.

Jolt Test MIL-STD-300

Jumble Test MIL-STD-301

Drop Test (8 ft) MIL-STD-302 (safe and operable)

40-Foot Drop Test (safe, no-fire)

3.6.3 Summary

We have considered the mechanical design details of a typical EED, the possible modes of failure of the EED when subjected to a severe shock and vibration environment, maximum vibration shock and acoustic noise data from flight testing, and also a comprehensive listing of test envelopes, and qualification and "off-limit" tests.

Having compared applicable acceptance tests and level flight data, we find that the environmental testing program developed for the Apollo Standard Initiator appears to be realistic. It has also been shown, however, that more severe environments exist, than are presently used for testing the ASI. Since the design and testing levels for an initiator must insure reliability, it is necessary that there be constant comparison of new flight data, and empirical scaling of such data for specific applications and revision of specifications as required. The testing program should include the combination of the most severe environments encountered by the EED, and a constant check on its performance both during and after exposure to these environments. A design and product improvement program for EEDs must include the use of failure data from off-limit tests (such as performed on the ASI) to determine the probabilities of failure if such environments are encountered, and methods of preventing such failures.

3.7 Prelaunch and Interior Conditions

3.7.1 Sterilization

Prior to the launching of a mission to any extraterrestrial object, a rigid sterilization process is applied to the missile, to prevent contamination of the target by organisms. The intent, as is well known, is to prevent interference, by imported life, with a subsequent biological study of the target. At first it may appear that explosive components could properly be exempted from the requirements, by reason of self-sterilization during explosive functioning. This is not the case: Incomplete combustion often occurs; possibly some devices are not intended to function except under certain conditions; some may be duds. The explosives must, for these reasons, be subjected to sterilization.

These are three major basic methods of sterilization - thermal, chemical, and radiation. Experience has already shown that space passage does little toward sterilizing a vehicle; neither the low temperatures nor the radiation are effective. Present concern is not with biological effects, but with the possibility of affecting explosives. Sterilization by gamma radiation may be dismissed at once; because of its inconvenience it does not appear to be a favored method for this application; furthermore, even if used, explosives, ceramics, metals, and plastic electrical insulation other than teflon are unimpaired by the sterilizing does of 10^7 Roentgen[33].

Chemical sterilization is often used. For purposes with which we are now concerned, two substances which have received attention as possible sterilizers are ethylene oxide and β-propiolactone.

Ethylene oxide is a colorless gas that liquifies at 10.8°C. It finds its major use today as an organic intermediate and it has been used as a fumigant to control pests in foodstuffs[34]. Its inflammability poses disadvantages (it is similar to ethyl ether) so that it is mixed with CO_2 to suppress its ignitability. In a closed vessel all mixtures greater than 3.6% of ethylene oxide with air, or ethylene oxide itself, can be ignited by a source of heat[35].

Ethylene oxide is a universal disinfectant, killing bacteria and other organisms effectively[36]. Being a gas, ethylene oxide has two distinct advantages: it is highly penetrating, compared to liquid, and it is effective at relatively low relative humidity (20 to 40% RH). The vapor will irritate the nose and eyes and concentrations of 50,000 to 100,000 parts per million will cause death in minutes. Maximum allowable concentration in air for people is 100 ppm. Many polymeric materials will sorb the gas, and hold it tenaciously, so that many hours of air-wash are required before sterilized materials can be safely handled[33]. In the case of space probes the ethylene oxide would be subsequently purged with nitrogen or helium to make a saving in the launch weight.

The sterilizing liquid β-propiolactone has a boiling point of 155°C and a freezing point of -33.4°C. It has a very irritating sweet odor and acts as both a lachrymator (tears) and a vesicant (blisters). It will react with water to form hydracrylic acid as well as with organic compounds having active hydrogen atoms, inorganic salts and some metals. Under conditions of maximum effectiveness (relative humidity greater than 75% RH) it is 25 times more effective than formaldehyde and 4000 times more effective than ethylene oxide[37]. Like ethylene oxide it is universal in its action on microorganisms but it is effective only on the surface. In applying it, it is often sprayed as a fine mist, at room temperature. It may be considered noncorrosive to metals for short exposures. If metals are exposed to the liquid for extended periods (months) at temperatures as high as 130°F then its use will be objectionable for metals such as brass, mild steel, 2024 aluminum, cast iron, red brass copper and galvanized steel. Under these conditions, 304 and 306 stainless steel, monel, and 1100 aluminum[33] would be very resistant. Prolonged exposure to the vapor will disintegrate nylon and polystyrene, and deform Mylar and polyvinyl chloride. In tests of β-propiolactone liquid in contact with plastics at 130°F Mylar was the only plastic that showed no apparent effect in 21 days[37].

Most other chemical disinfectants can be eliminated from this study because they lack the broad spectrum of lethal action obtainable with ethylene oxide or β-propiolactone or for other limitations such as their corrosive nature, undesirable residues, or insufficient data concerning their behavior.

The sterilization problem is compounded by the fact that clean surfaces are not sufficient; microbes within the materials must be killed. Studies made at Fort Detrick by Dr. Phillips and his group concluded that electronic components were contaminated with imbedded microorganisms. The use of dry heat, ethylene oxide and gamma radiation has been recommended by NASA and JPL[38].

Dr. G. Hobby, in charge of sterilization at JPL, has repeatedly urged that space probes subsequent to Ranger be designed from inception with cognizance of the restraints imposed by sterilization[33]. He has further stressed that heat sterilization (125°C for 24 hours) be used wherever possible[39]. The chief advantage of dry heat as a sterilizer is that it kills internally trapped microorganisms.

In summary we expect that the sterilization process will impose more severe restrictions on the design of the EED than the effects to be expected from the microorganisms themselves. Since it seems very desirable to use heat soaking to kill organisms throughout the bulk of material, the explosives selected must be relatively unaffected by the temperatures required.

The use of sterilizers other than heat does not appear to present serious difficulties; it is recommended, however, that the use of any organic materials used as sealants, gasketing, or electrical insulation be reviewed with respect to microbial attack and their resistance to any sterilizing process that may be used. Although recent research[40,41,42] shows that the emphasis is on heat sterilization it would be advisable to assume that several methods of sterilization may be used to cope with various situations. There may even be the necessity for sterilization near the end of the return journey from a planet while the probe or craft is in its quarantine period about the earth.

Table 9

DRY HEAT STERILIZATION[43]

Temperature °C	Temperature °F	Time Minutes
170	340	60
160	320	120
150	300	150
140	285	180
121	250	overnight

Figures based on survival times of the more resistant bacterial spores.

3.7.2 Effect of Microorganisms

The previous section had to do with the effect of sterilization processes. But in the case of manned flight, obviously sterilization can not be complete. In any event, consideration must be given to the possibility of some organisms remaining within the vehicle, whether they be bacteria, yeasts, fungi, or algae, is with the manner they themselves can affect explosive devices or the effect on these devices resulting from the sterilization techniques used to eliminate the microbes.

All microbes require a source of carbon, nitrogen, trace elements, energy, and something to accept the products of their metabolism. The conditions required for some microbes to multiply are very restrictive. Many varieties can be cultured only within narrow temperature limits. Slow freezing or high temperature (See Table 9) will kill most of them. Quick freezing merely inactivates them. A narrow range of hydrogen ion concentration and the presence or complete absence of oxygen may be necessary for the survival of some types. There are also some microbes known as lithotrophic or "stone eating" types that can live in environments devoid of organic matter since they can use carbon dioxide as their source of carbon and they can oxidize materials such as sulfur, ferrous iron, hydrogen sulfide and ammonia[44].

Russian researchers[45] have shown that bacteria can act as biological depolarizers and activators in the process of electrochemical corrosion in sea water. Sulfate-reducing bacteria growing on steel in sea water containing organic substances and sulfates can serve as depolarizers, activating the the oxidation of hydrogen which accumulates on the cathode of microgalvanic elements during the electrochemical corrosion of steel. Carbon dioxide, accumulating due to saprophytic bacteria and as an end product of organic substances, destroys the protective properties of the surface layer of steel. Its activation causes a sharp shift of electrode potential to the negative side and creates conditions for a more rapid development of the microgalvanic pair that enhances the self dissolution of the metal. Stainless steel weight losses of 0.002% in one year have been measured and up to 9% for carbon steel with certain saprophytic bacteria.

There has been research on biological fuel cells which shows that certain strains of bacteria can reduce carbonaceous fuels to hydrogen sulfide, which can dissociate at an anode-electrolyte interface. A prototype made by Dr. F. D. Sisler of General Scientific Corp. used an active culture of desulfovibrio to reduce formaldehyde in the presence of sulfuric acid. Oxygen and water were used at the cathode to complete the cell.

From the foregoing it is apparent that organic, inorganic, and metallic substances can be affected by microbial action to some degree under suitable conditions. With the metals and most inorganics we expect that the effects will be negligible. The organic materials which may be used should be scrutinized on an individual basis to determine their suitability.

It is our opinion that direct bacterial action will afford much less cause for concern than the sterilization procedures, discussed in Section 2.7.1.

3.7.3 Biological Wastes

Biological waste products from astronauts will probably be contained or converted and hence are expected to pose no problem. It may be advisable to use treatments that will inhibit the growth of bacteria or fungi that can lead to the corrosion of metal parts. The chief waste products are listed below.

Feces

Important components are hydrogen sulphide, methane and ammonia. The materials that could be affected include aluminum, copper, unalloyed steel and epoxy moldings.

Urine

Aluminum, magnesium and titanium react with urine components. the main problem is ammonia gas. Stainless steel and flurocarbon gaskets are recommended for processing containers.

Flatus

An important component is hydrogen sulphide, which affects aluminum and copper as well as lubricants.

Vomit

This contains a corrosive, hydrochloric acid, in 0.1% to 0.5% concentration.

Perspiration

Salt content is usually quite large.

Fungi and Bacteria

Almost any material containing organic compounds in trace amounts can support growth of some microorganisms which can lead to corrosion of metal parts.

3.7.4 Other Chemical Considerations

From the chemical viewpoint it appears that there will be no difficulty in using materials that have been found useful for aircraft and missiles in spacecraft applications. There are a few exceptions, mostly polymers, thermal control paints, or semiconductors susceptible to radiation damage. Although a kaleidoscopic array of chemical species may be encountered from launch to return, the effective concentration of the reactants will be relatively small for most materials of interest.

The evaporation of materials will not be serious for extended voyages except possibly in the case of alkali metals, or plastics with volatile plasticizers. The substitution of pure polymers for the latter will probably be acceptable if other properties are suitable. The reduced pressures of outer space will also tend to remove the gaseous layers which may be adsorbed on material surfaces. This removal of gas may inhibit crack formation, although some research indicates contrary results.

The effect of various gases which may be encountered on a mission was discussed in the second progress report on this project. To summarize the findings, they are expected to be harmless, because of low pressure or innocuous nature or both. There are possible exceptions, notably Venus, where the atmospheric pressure may be high. If the equipment is suitably designed to withstand the pressure, it will in all likelihood withstand any chemical attack, at least for the required duration.

3.7.5 Prepulses

Ideally, the only electrical energy which should reach the bridgewire of an electroexplosive device is the firing pulse. In practice, however, there may be other electrical stimuli which will reach the bridgewires of these devices; for example, continuity checks and in-flight monitoring pulses. Some less obvious sources of electrical energy to the bridgewire are induced currents which may arise from electrostatic and electromagnetic fields and thermo-electric currents generated by dissimilar metal joints subjected to temperature gradients.

The effects of such prepulses on the initiator can vary, depending upon the nature of the prepulses and the construction of the initiator. On a microscopic scale, raising the temperature has the immediate effect of increasing the sensitivity until sufficient decomposition occurs to cause desensitization as a result of chemical and physical changes, if the energy input is less than sufficient to cause detonation.

The indirect effects of prepulses are of two types. In the first type, the decomposition of the surrounding explosive may produce a change in the surface coefficient of heat transfer between the wire and the explosive. An actual void may be produced, or the decomposition products may simply exhibit a new transfer coefficient. In either case the sensitivity of the initiator will be changed[46] and usually reduced. If a sufficient number of prepulses of the proper magnitude are applied some hot wire systems may be completely desensitized[47]. In the second type heating the wire with a prepulse may produce a slight axial expansion of the wire causing it to bow. It is possible under this condition to create a void which could alter the sensitivity of the hot wire initiator.

It is possible that these effects will be less in EBW initiators, since decomposition due to prepulses is local while the action of the exploding wire on the surrounding explosive may be strong enough to overcome local desensitization. It should be noted, however, that many so-called EBW initiators that rely on a thermal stimulus in addition to shock fall short of being true EBW's; they would therefore be expected to react similarly to hot-wire initiators.

The magnitude of the prepulses can be reduced by shielding the circuits from stray fields and by limiting pre-flight checkout and in-flight monitoring signals. The effects of prepulses can be reduced by selecting initiator constructions which employ heat-stable explosives in conjunction with sound component configurations. In general there is a lack of data concerning the effects of prepulses, and this is an area which should be further investigated. It should be apparent that prepulsing is not a circumstance peculiar to space missions, but is one which may affect all phases of electric ordnance. However, because of the repetition of tests and checks, there may be a tendency toward greater exposure of the devices when installed in a space vehicle. The selection of initiators must be correlated with expected test procedures, in-flight monitoring checks, and suspected stray currents from thermoelectric, electromagnetic, or other sources.

4. CHARACTERISTICS OF EXPLOSIVES

To aid in the selection of suitable explosives, the tables below may be useful.

4.1 Effect of Radiation

Explosive materials are by nature metastable. In spite of this they seem to be quite resistant to damage by ionizing radiation. The threshold of damage for most explosive materials is above 10^6R.

Table 8 is taken from work reported by Kaufman[48], which shows the gas evolved in 40 days of irradiation from a .41 MeV source whose average radiation is 10^5R per hour for a total exposure dose of about 10^8R.

TABLE 10

IRRADIATION EFFECTS ON EXPLOSIVES

Explosive	Gas Produced in cc/gram
Lead Styphnate	.07
TNT	.09 - .10
RDX	1.49
PETN	2.42
Lead Azide	3.5 - 3.92
DDNP	5.6

Postradiation gas evolution was noted for RDX and lead azide, indicating that sustained decomposition had been started in these explosives.

In order to determine the effect of radiation upon an explosive element the total expected exposure dose must be accumulated using the time and expected radiation flux, and this figure compared with the radiation characteristics of each explosive material. Shielding of the explosive and the temperature conditions for the entire time before functioning of the explosive must also be considered.

Radiation protection for explosive elements is largely a question of shielding. The initiator, a critical part of the explosive train should be easy to protect from radiation by shielding because of its small size. The larger charges of high explosives, pyrotechnic mix, or propellants are much more difficult to shield, but may have less need of it. It is likely that these substances are more resistive to radiation damage or because the outermost particles tend to shield the others the expected damage can be practically offset by a small increase in the quantity of explosives.

4.2 High Temperature and Vacuum Effects on Explosives

The high vacuum of space can cause high evaporation loss in a good many explosive materials. This is minimized or entirely eliminated if the explosive is hermetically sealed in its case, a procedure at present called for the Apollo Standard Initiator. This type of seal is of course, more difficult with large solid propellant charges. It appears that it would be possible to test the effects of high vacuum on large solid propellant units if this has not already been done.

Data on vapor pressure or evaporation rate is not readily available on most explosives and what is available may not apply to space vacuum conditions. Data from vacuum stability tests shown in Table 11, used to evaluate the temperature stability, is available on most explosives but the vacuum used comes only within about 5 mm of mercury of space vacuum conditions.

High and low temperature extremes are the most important environmental conditions that explosives will encounter in space. Temperature has a direct effect upon the rate of chemical reaction. An explosion is an extremely rapid chemical reaction; and it is reasonable to suppose that the low temperature of space can also cause problems by slowing the rate of explosive reaction, by reducing the output, and by causing thermal stresses which could crack the explosive. Fracture of the explosive mass usually increases the rate of reaction by exposing more area to hot burning gases in the case of a solid propellant, or it may reduce the output by producing a discontinuity in the explosive train.

Long time storage at low temperature would retard decomposition reaction in explosive materials, thereby prolonging storage life.

TABLE 11

HIGH TEMPERATURE AND VACUUM EFFECTS ON EXPLOSIVES

	Explosion Temp °C	Wt. loss percent for 48 hrs at 100°C	Vacuum* stability cc/gm/40 hr at** 100°C
Nitrocellulose (12.6%)	170 d	-	1.0 (11. at 120°C)
DDNP	195	2.10	7.6
PETN	225 d	.10	.5 (11. at 120°C)
RDX	260 d	.04	.7
Lead styphnate	282	.38	.4
Lead azide	340	.34	1.0
Ammonium perchlorate	435	.02	.13
TNT	475 d	.20	.10

d decomposition
* 5 mm Hg
** Tested in one gram samples for initiating explosives and 5 grams for others.

Several secondary effects of high temperature on explosives are of importance and must be considered. Chemical compatibility reactions, certain radiation decomposition reactions and the rate of evaporation under space vacuum will increase with increased temperature.

In Table 11 the first column gives the temperature which will cause explosion or decomposition in 5 sec. The next two columns show respectively the percentage decomposition at 100°C on a weight loss basis and the volume of gas evolved under vacuum (a test of stability). The actual operating high temperature limit would be somewhere below the 5-second explosion temperature in column one, depending upon the time-temperature characteristics and other conditions which may affect decomposition, such as compatibility.

A great deal of data are available for explosive ordnance systems for temperatures of 160°F (71°C). This may not be a high enough temperature for space application. Estimates of temperature extremes for a Lunar or Mars mission (see Figures 2-1, 2-2 of report P-B2333-4) give a range of -234°F (-148°C) to 315°F (157°C) for up to 250 days. This high temperature, sustained for more than an hour, is just within the present capability of special high temperature explosive materials. The Venus mission (Figure 2-3 of report P-B2333-4), with a high temperature in the vicinity of the planet of 800°F (427°C), is out of the question with present explosives. However for very short times, insulation may make it possible to use explosives under these conditions.

Explosives systems intended to function at 315°F (157°C) must be tested for functioning at or after exposure to this high temperature, depending upon the requirement. To the best of our knowledge almost no explosive component is tested for functioning at the high or low temperature extremes. The 160°F high temperature test pertains to storage conditions, and the functioning test was made at ambient room temperature. If it is required that the space explosive device operate at low or high temperature extremes, the firing tests used to determine the sensitivity and output of the explosive element should also be made at these temperatures.

Considerable work has been done to improve the temperature sensitivity of explosive elements. Much classified work has been done to develop new high temperature explosives, and to find means of initiating them without using the temperature-sensitive primary explosives.

In general there is a great deal of information available on the use of explosives under various environmental conditions. The missing information in this field seems to be at the extreme environmental conditions and more specifically at long time periods at these conditions. These environments are temperature, radiation, and space vacuum. Also combinations or interactions of these environments may be important and need additional research to define their effects more clearly.

5. CONCLUSION AND SUGGESTED FUTURE STUDIES

The science of space environment is a new and rapidly expanding field. Even as this report is being written new data are being generated. It is conceivable that within a year's time most of these data will be out of date. This, of course, suggests that a continuous monitoring of the research projects be done. Keeping this in mind, we have prepared a summary of the data obtained during the course of this survey.

a. Temperature Pressure Duration Environments

A study of the profiles listed in the report indicate that the most hazardous environment in terms of explosive initiators would occur on entering the atmosphere of Venus. The high temperatures which might be encountered are beyond the ability of present explosives to withstand. Furthermore, the pressure which might possibly exist near the surface of Venus is of a magnitude for which initiators are not normally designed or tested. Performance at such pressures could be checked rather easily, however. While the time spent in such an environment would probably be relatively short on early missions, the temperatures are so high that one must either find means other than explosives to accomplish necessary tasks, develop new explosives or protect the initiators from the temperature.

A somewhat less conspicuous problem could occur in passage between the planets. In this case extremely low pressures and both very low and very high temperatures exist for extended periods of time. While, generally speaking, there are explosive devices capable of withstanding these pressures and temperatures we are approaching marginal conditions, and explosives may be subjected to such environments for many many months before being expected to function. Extensive tests of any final design are indicated.

The welding effects that occur as a result of the extremely low pressures occurring must be kept in mind wherever the explosive device performs a mechanical function such as in a piston motor for example or whenever the explosive initiator is in a mechanical safety device such as an out-of-line rotor.

b. Electromagnetic Radiation Environment

A study of the literature indicates that the most hazardous source of natural electromagnetic radiation in space is the Sun. An analysis of the solar spectrum has been completed from low frequency radio waves though x-rays and all indications are that levels are such as to constitute no serious hazard even for an extended period of time.

The most serious problem is that of temperature control of the vehicle due to absorbtion in the ultraviolet infrared band. Excellent progress has been made in this area. During the writing of this final report, a text book on theory and methods of obtaining this temperature control was published and is included in the bibliography even though we do not reference it in the text [49].

c. Other Electrical Environments

One of the potential hazards to explosives in spacecraft is presented by the electrical discharges that might occur as a result of lightning or electrostatic charge build-up. In general it would seem that the lightning hazard, at least in earth atmosphere, could be minimized by choice of launch time and proper protection of the launch site. However,the possibility of lightning in the atmosphere of other planets or even during reentry into earth's atmosphere must be considered since in these instances one may have no control over the temporal factor.

The electrostatic charge build-up problem is one that may be with the spacecraft under any conditions. It is pointed out in this report that the amount of spark energy required to initiate a given explosive is also a function of the amount of series resistance in the initiating circuit and that the values of resistance at which some explosives appear to require minimum energy are of the same order of magnitude as might be expected from leakage paths on insulators.

Because of this there can be considerable risk in extrapolating the data from a simple circuit laboratory test of initiator electrostatic sensitivity to all of the installations on a spacecraft. Careful and complete testing of the final design is indicated as well as designing of the initiator and associated circuits to minimize any possibility of firing from an electrostatic source.

d. Particle Environment

The Van Allen belt and our Sun both serve as rich sources of relatively high energy electrons and protons. Fortunately present information on explosives indicates that the common ones are relatively insensitive to this radiation; however, intensive irradiation can only degrade the explosives. Of more immediate concern are the other circuit components that may make up the firing circuit. Transistors, semiconductors, and similar components have been shown to be quite sensitive to such radiation, and some plastics can have their properties completely changed by it. The materials used in the initiator should be chosen from those that show very little effect. Whenever possible proper radiation shielding should be incorporated. Cosmic rays have been only lightly considered; these should be studied further.

The larger particles such as meteoroids, asteroids and solar dust are generally of such a size that should a direct hit on an initiator occur, the initiator could be completely destroyed. However, the probability of this occurring is apparently quite small and probably grows even smaller as the particles become larger and therefore more damaging. It is also possible to minimize the likelihood of such collision by proper choice of spacecraft course, and it is assumed that such procedures would be followed, particularly for a manned spacecraft. This is a case, however, where redundant circuits may be an added safeguard.

One special case must be kept in mind. This has to do with meteor showers. At least two craft ceased operation after encountering such swarms. Astronomers have catalogued and can predict the location of many such showers, so that they can be avoided, but we cannot be certain that there are not others in space still unknown. More information is needed regarding the actual effect of these showers on spacecraft if protective means are to be employed.

e. Chemical Environment

The possibility of chemical changes occurring in the initiators as a result of particle irradiation has been subjected to some study and it was concluded that it was probably of not much concern. In addition, on-board chemical environments such as produced by the men themselves or by functions of the spacecraft also appear to be only a minor hazard. Some additional analysis of this problem and the chemical environments represented by the atmospheres of other planets is indicated and should be considered when more information on the atmosphere becomes available.

One potentially serious problem has been uncovered, which has to do with microorganisms and sterilizing processes. In general the microorganisms themselves would not seem to offer any serious problems except on the longest of missions where some effects might be possible. The sterilizing procedures being used to eliminate these microorganisms is of more immediate concern, and while no specific problems are defined, the temperature and chemicals used should be carefully studied in terms of their possible effects on explosive initiators.

Another factor that should be considered is the incorporation of sterilization procedures in each step of the initiator construction so that the final device is already up to standards.

f. Shock and Vibration Environment

This is a region in which specific data is still being compiled and not all parameters have yet been defined. It would seem that the best immediate course is to continue to apply present specifications and tests to initiators until more information is obtained.

In summary the study has indicated that there is no environmental problem so acute that instantaneous action is required. This is generally borne out by the success of our present mission in space. However, there are several areas which should be investigated and suitable steps taken to insure the safety and reliability of explosive devices as space missions become more extensive and complicated. An orderly program of investigation and correction is indicated.

It should be reemphasized that while the statements made in this report are accurate within the present state of knowledge, we are continually learning more and more about space environment, and the operation of explosive devices in such environments should be continually reviewed as our knowledge improves.

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Good reference on space material-section on explosive etc.

STL Shock Isolation Panel - Ground Shock Problems in Hardened Missile Installations, AD 337 694 FIL Log No. 1641 (Confidential), April 1960, Space Technology Laboratories Inc., Los Angeles 45, California

Prevention of damage to missile installation due to the ground shock induced by a nuclear burst. Displacement and accelerations of magnitude greater than that listed in MIL STD 810, but effect on missile itself negates consideration of lesser effects on EEDs.